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FOREWORD

General

1. The purpose of the Aircraft Accident Digest is to disseminate accident report information to all Contracting States. Publication of the Digest began in 1951. Over the years States have reiterated their interest in the Digest not only as a valuable source of information for accident prevention, but also as a training aid for investigators and educational material for technical schools.

Selection of Accidents

2. The Digest contains accident reports which have been selected by the Secretariat from those sent by States. Reports were selected on the basis of:

a) their contribution to accident prevention; or

b) the successful employment of useful or effective investigative techniques; or

c) number of fatalities.

It should not be seen as being statistically representative of the world distribution of accidents.

3. The Final Reports are usually published as received. However, lengthy reports may be abbreviated by omitting from the "Factual Information" any information which is also detailed in the "Analysis", or by omitting Appendices.

States' Co-operation

4. States are encouraged to send to ICAO those Final Reports which meet the criteria in para. 6.14 in Annex 13. The reports must be submitted in one of the working languages of ICAO, and in the format presented in the Appendix to Annex 13. Thus they contain the following parts: Factual Information, Analysis, Conclusions and Safety Recommendations. The accident cause(s) are included in the conclusions.

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AIRCRAFT ACCIDENT REPORTS AS RECEIVED FROM STATES

No. 1

Boeing 747-136, G-AWNP, accident at Grantley Adams International Airport, Barbados, on 28 December 1978. Report dated 15 December 1980 released by the Civil Aviation Division, Barbados.

SYNOPSIS

The accident was notified to the State of Registry, United Kingdom on December 29, 1978 by the Technical Director - Aviation, Barbados, on behalf of the Ministry responsible for Civil Aviation in Barbados.

The investigation was carried out by the Technical Director - Aviation on behalf of the Civil Aviation Authorities in Barbados. He was assisted by a team of experts which included personnel from British Airways and Shell Antilles & Guianas Ltd. Certain aspects of research and consultation took place with the Accident Investigation Division of the Department of Transport, Canada.

Shortly after the aircraft had arrived on the apron at Barbados and passengers had commenced disembarkation, a refuelling pipe coupling swivel assembly became disconnected resulting in a fuel spillage which became ignited. The remaining passengers were evacuated via the passenger steps and two door slides. Eleven passengers and a refueller operator sustained minor injuries. The fire, which was extinguished by the Airport Fire Service, caused extensive damage to the aircraft.

Note.- All times in this report are GMT.

1. FACTUAL INFORMATION

1.1. *History of the Flight*

The aircraft was operating a scheduled service Flight BA-251 from London to Port of Spain via Antigua and Barbados. The aircraft departed London on the 28th December at 1000 hours, landed at Antigua at 1844 hours and departed for Barbados at 1948 hours. The flights were uneventful and the aircraft landed at 2042 hours. It taxied onto Gate 'A', was parked at 2046 hours and the engines were then shut down. Mobile passenger steps were positioned at Nr 1 and Nr 2 left doors and the passengers started to disembark. At this time a Shell refuelling tanker, local registration A91, was positioned under the starboard wing just outboard of Nr 3 engine, parallel to the fuselage and facing in the same direction as the aircraft. Two fuel supply hoses were connected, by a Shell refueller to the aircraft couplings to refuel the aircraft with Jet A1 fuel. Refuelling was commenced by a Shell supervisor and as the fuel pressure increased the quick disconnect swivel assembly of the inboard coupling parted from the controller. The refuelling hose fell towards Nr 3 engine and discharged fuel onto the engine and the ground. A fire broke out immediately. The Supervisor stopped the fuel flow by means of the 'deadman control'. Approximately 72 U.S. gallons of fuel were discharged before the fuel flow ceased. The refueller sustained minor injuries jumping from the tanker platform.

The crew, who were still in their seats, reported that at 2049 hours a wheel well fire warning occurred so the Captain informed the Air Traffic Controller of a fire. Flight deck to ground communication had not been established at that time. Shortly afterwards, Nr. 4 engine fire fault indication light came on but the fire bell did not ring. Nr 4 engine fire drill was carried out. Smoke was observed drifting from right to left and the Captain then ordered an emergency evacuation. The "passenger evacuation land drill" was carried out and the passengers and crew evacuated the aircraft using passenger steps which were already in use at Nr 1 and Nr 2 left doors and the slides at Nr 1 right and Nr 5 left doors. Air Traffic Control reported seeing smoke at 2051.

The Airport Fire Services arrived quickly on the scene and the fire was extinguished approximately ten minutes after it had started, by the use of foam, BCF and CO_2.

1.2 *Injuries to Persons*

Injuries	*Crew*	*Passengers*	*Others*
Fatal	–	–	–
Serious	–	–	–
Minor	–	11	1

1.3. *Damage to aircraft*

There was severe fire damage to the following areas:-

Starboard wing to fuselage fairings.

The leading edge of the starboard wing between fuselage and Nr 3 engine pylon, including leading edge flaps and mechanisms, wiring looms, pneumatic

ducting and fuel valve actuators. Approximately six feet of the front spar web was distorted due to heat damage.

Nr 3 engine inboard side and cowlings and Nr 3 engine pylon.

The starboard wing landing gear and doors.

The starboard inner aileron.

The mid and aft sections of starboard inner trailing edge flaps and Nr 5 and 6 flap canoe fairings.

1.4. *Other Damage*

The Shell "Oxford" fueller and trailer, airport registration number A91, had burn damage on the left side and the fueller left side rear wheel tyres had been burnt and deflated. The fuel hoses and control systems were substantially damaged by fire.

A toilet servicing truck, airport registration number A34, was superficially burnt on the bonnet, the front and right side. The front right tyre had burn damage. This vehicle was removed from the accident site shortly after the accident without proper authority.

1.5. *Personnel Information*

The cockpit crew held valid and appropriate licences for the performance of their duties and the cabin staff were appropriately trained for the performance of their respective duties.

The staff employed by the fuel supplier were also well trained and had long experience. A comparison with refuelling procedures and regulations at three other International Airports showed that an accepted international standard was in effect at Grantley Adams International Airport.

1.6 *Aircraft Information*

The aircraft registered in the United Kingdom G-AWNP held a valid certificate of airworthiness and was maintained in accordance with a maintenance scheduled approved by the United Kingdom Civil Aviation Authority. The aircraft was being fuelled with Jet A – 1 fuel.

1.7 *Meteorological Information*

The wind speed and directions at the time of the accident was 110^0 – 15 Kts. Temperature 27^0C.

1.8 *Aids to Navigation*

Not applicable

1.9 *Communications*

There was no communication between the cockpit and the ground as two-way contact was not established on arrival of the aircraft.

1.10 *Aerodrome Information*

Not applicable

1.11. *Flight Recorders*

These were sent to the United Kingdom for analysis to assist in the investigation.

1.12 *Wreckage*

Not applicable

1.13 *Medical*

Medical records indicated that eleven passengers and one refuelling operator were treated for minor injuries.

1.14 *Fire*

The fire which occurred following the spillage of fuel into the number three power plant was quickly extinguished by the Airport Fire Service using FP – 570 (Foam) 002 and B. C. F.

The fire call was raised first by the alert observation of the Fire Service Control Room attendant. There was full attendance and no operational difficulties were encountered. The Fire Service personnel were well trained and carried out their task in a highly professional manner and averted what could have been a major disaster.

1.15 *Survival Aspects*

Local Ground Handling staff met the aricraft as usual and the doors were opened to permit disembarkation. Passenger steps had been positioned at Nr 1 and Nr 2 left doors.

Fifteen first and twenty four economy class passengers had cleared the aircraft using Nr 1 left door when smoke was observed in the vicinity of steps at Nr 2 left door by a stewardess, who had followed the passengers down. The smoke was also seen engulfing these steps by another stewardess located at Nr 2 left door who then redirected passengers to Nr 1 left door. Approximately 90 passengers left the aircraft using the steps at Nr 1 left door after the steps at Nr 2 left door were obscured by smoke.

The opening of Nr 1 right door was delayed due to the presence of smoke on the right side of the aircraft but it was subsequently set by one of the cabin attendants, it opened and the slide deployed. Difficulty was experienced in opening the door and it was reported by the cabin staff concerned that they thought the power assist did not function as it required two stewards to complete the opening. Approxi-

mately 25 passengers, three cabin staff and the Engineer Officer left the aircraft using the slide at Nr 1 right door.

Nr 4 left door was not opened because smoke and flames were observed through the window; Nr 3 right and left, and Nr 4 right doors were not used due to the presence of smoke and flames observed from the cabin.

Nr 4 left door was opened by a catering loader but shortly afterwards smoke entered the door and the catering truck withdrew with the alert attendant closing this door. The door was not used in the evacuation.

Prior to the evacuation signal, a passenger reported to a stewardess adjacent to Nr 5 right door that he thought an engine was on fire, smoke and flames were observed through the window by the stewardess so she moved to Nr 5 left door. On the receipt of the evacuation signal this stewardess put Nr 5 left door to automatic and opened the door. The slide deployed and approximately 120 passengers and three cabin staff left the aircraft using this slide.

Eleven passengers sustained minor injuries as the result of using the slides and were treated by a doctor in the Airport Medical Unit.

The majority of the joining cabin crew were on the apron in the immediate vicinity of the aircraft at the time of the accident. A number of this cabin crew and staff from Seawell Air Services Ltd. assisted in helping passengers off the slides and directing them away from the aircraft. The Captain and Co-pilot evacuated the aircraft via Nr 1 left door.

The Captain and the Co-pilot of the joining crew were in the aircraft lounge when the fire started. They evacuated the aircraft via steps at Nr 1 left door and then assisted at the bottom of the slides.

Most of the cabin crew involved in the accident reported on the large amount of hand baggage carried and the difficulties experienced in separating the passengers from their hand baggage and their duty free purchase prior to going down the slides.

Some passengers did succeed in taking duty free bottles down the slides and as a result a number of bottles were broken around the base of the slides and created a hazard during the evacuation.

The evacuation of the aircraft is estimated to have taken about three minutes.

1.16 *Tests & Research*

Examination of the starboard inboard swivel coupling which had become detached showed that it was in the locked condition. However, while the surface was entirely blackened with a sooty deposit from the fire there was a clean scrape mark across the top of the locking latch which suggested that the coupling had been rotated to the locked position after the fire. There was no mechanical defect in the swivel coupling or the nozzle to which it was attached prior to the accident and when the

coupling was refitted to the nozzle and correctly locked, the assembly withstood a test pressure of 90 psi without leakage. Subsequently, after various other pressure tests had been carried out with the subject assembly, a new hose was fitted to the assembly and a pressure test of 300 psi was carried out without leakage.

To establish that the specific fueller nozzle/aircraft coupling combinations did not leak under fuelling conditions, the two nozzles, from the "Oxford" in use at the time of the accident were disconnected at the quick disconnect/swivel assemblies and fitted to a "Somerset" fueller. The nozzles were then coupled to their original positions on the right wing of the aircraft. Refuelling pressure was applied and no external leakage was observed.

To establish that the aircraft fuel system was free from leaks, the right hand centre wing tank jettison valve was manually selected open on the over-ride handle in order to pressurise the engine crossfeed system when the refuel manifold was pressurised. When normal refuelling pressure (45 psi) was applied, from a "Somerset" fueller, firstly to the left wing refuelling points and secondly to the right wing refuelling points, no fuel leakage was observed from the aircraft apart from a small leak from the right hand jettison nozzle outlet at the wing tip.

The fuel tanker involved in the accident had been checked for filter contamination on the 27th December and this check involved detaching the hose at the swivel coupling. The tanker had not been used for any fuelling operation between the filter check and the time it was connected to G-AWNP on the 28th December.

The Flight Refuelling quick disconnect swivel coupling is designed to connect the fuel hose to the controller section of the fuel coupling assembly. The swivelling feature allows the controller to be manipulated during coupling and uncoupling, to and from the aircraft, without twist having to be imparted to the hose. This feature is achieved by an outer lock ring, carrying an internal cam ring which in the locked position engages 6 ball bearings in an annular groove in the controller. The quick disconnect feature, which provides easy access to a conical filter inside the coupler, is achieved by turning the outer lock ring and hence the cam ring release the balls, thereby, allowing the coupler and hose to be removed from the controller. This turning the outer ring in one direction locks the coupling and the opposite direction unlocks the coupling.

Movement from the locked to the unlocked position is prevented by two mechanical levers both of which have to be manually depressed to allow the outer ring to be turned to the unlocked position.

As the evidence indicated that the Flight Refuelling quick disconnect swivel coupling became detached probably because the locking device was not in its fully locked position at the commencement of fuelling, tests were carried out on a typical coupling. The tests showed that the Flight Refuelling quick disconnect system will not separate, even up to pressures of 300 psi, when it is locked in the closed position. It was also shown that, when not locked, there is a range of positions for the locking levers, when the coupling is mechanically linked and will withstand fuelling pressure without leaking. In all tests carried out on the un-

locked coupling a rotational movement occurred towards the unsafe more open position and an unlocked coupling could be set up to come apart during simulated fuelling runs on the Fuel Depot test rig.

When the above test were repeated on the subject quick disconnect coupling the results obtained were similar to those for a typical coupling.

Nr 3 engine was removed and sent to the engine manufacturer for investigation and repair. The manufacturer has advised that the engine fuel system was free from leaks. There was no evidence of any residual fire in the tailpipe and the Nr 4 bearing compartment exhibited no distress that would have been contributory to a residual fire in aft engine areas.

The manufacturer further advised that typical engine temperatures three minutes after shut down are as follows:-

6th Stage Disc	606%F
66th Stage Blade Platform	615%F
In Gas Path area	617%F
Turbine Exhaust Case ID	617%F
Turbine Exhaust Case OD	597%F to 607%F

Investigation into the power assist system of Nr 1 right door failed to reveal any defect to account for the reported difficulty to effect the opening.

The reason for the fire warning indication on Nr 4 engine reported by the crew is attributed to wiring damage by fire in the wing leading edge.

1.17 ADDITIONAL INFORMATION

1.17.1 *Investigation*

The personnel from British Airways and Shell Antilles & Guianas Ltd. arrived in Barbados on 29th December 1978 and the Investigation Team, as a group, first examined the aircraft on the evening of 29th December. The "Oxford" fueller and trailer were still in position under the starboard wing but the platform had been lowered. The open end of the right hand deck hose was lying on the platform floor pointing in the direction of Nr 3 engine. The fueller delivery meter indicated that 72 U.S. gallons of fuel had passed through it. The accumulative readings for this fueller were subsequently checked by the refuelling company and the 72 U.S. gallons associated with the accident were confirmed.

The aircraft starboard inboard refuel coupling still had the fueller right hand deck hose nozzle controller attached to it. The nose having separated from the controller at the quick disconnect swivel coupling. Both the controller valve and the associated aircraft refuel valve were in the open position. The wire mesh filter which is normally contained in the hose swivel coupling was found on the ground under the centre line of the aircraft and in line with the position of the disconnected hose indicating that it had been ejected by the flow of fuel when the hose became detached.

The aircraft starboard outboard refuel coupling seal plate was free of soot, indicating that disconnection had taken place subsequent to the accident. The blanking caps were hanging loose and were covered with soot. The associated aircraft refuel valve was in the shut position.

The fuelling isolation valve actuator in the right hand wheel well was indicating open as would have been necessary for a single side refuelling. The sooting pattern on the rear spar showed this to have been the selection at the time of the fire.

All tank refuelling valves were found selected shut at the refuel control panel on the left hand wing.

The engine fuel feed valve actuator for each engine indicated shut.

Nr 3 engine fuel condition motor mechanical indicator was at cut-off i.e., shut.

All fuel system selections on the flight deck were normal with respect to the required drills, except that the reserve tank transfer valves were found selected open. This would have permitted gravity transfer of fuel from the reserve tanks to the outer main tanks had the reserve tanks been refuelled. The transfer would have been entirely within the fuel tanks with no external connections.

All five fire handles, one per engine and one for the APU, had been pulled. The external thermal discharge indicator discs were all intact including those on the outboard side of Nr 3 pylon.

All eight engine fire bottles and the APU bottle were removed from the aircraft and were found to have been discharged electrically, i.e. all squibs had operated and the normal discharge diaphragms had been fractured. The thermal over pressure diaphragms were intact.

1.17.2 *Analysis*

There is no doubt that from all the evidence seen and from subsequent statements from all concerned that the Oxford fueller right hand deck hose to the inner aircraft coupling, separated at the quick disconnect swivel coupling under fuelling pressure. The resultant stream of Jet A1 was probably ignited spontaneously by impinging on the hot engine.

Evidence indicated that for some reason the locking device was probably not in its fully locked position at the commencement of fuelling. The design of the locking device preludes possibility of accidental unlocking, therefore an unintentional human error in the inspection of handling of the refuelling equipment is probable.

Fuel would have continued to flow from the open ended disconnected hose and fuel would have flowed through the open aircraft coupling from the other hose until pumping stopped. From start up to shut down was a matter of seconds and the fueller meter reading supported by records of the totaliser shows that 72 American Gallons were passed. In tests at the Shell Depot witnessed by British Airways and Ministry of Civil Aviation, it was clearly demonstrated that the

Flight Refuelling type quick disconnect system will not separate even up to high test pressures provided it is locked in the closed position. It was also clearly demonstrated that although not locked there is a range of positions for the locking lugs in which the coupling is mechanically linked and will withstand fuelling pressure without leaking. In all the tests done on unlocked couplings the movement was towards the unsafe more open position and an unlocked coupling could be set up to come apart during simulated fuelling runs on the depot test rig. It is significant that in one test starting with coupling unlocked but just mechanically linked the coupling parted, the hose dropped approximately two feet to the concrete yard and immediate examination showed that the interlocks had moved to the locked position as a result of the fall. This may explain the fact that the interlocks appeared to have been moved to the locked position after the fire.

All the above tests are a confirmation of the basic integrity of the Flight Refuelling quick disconnect coupling which has been used for many years in hundreds of thousands of fuelling operations without previous incident.

As indicated, personnel of the fuel supplier were properly trained and had received recent refresher training. In addition the procedure and regulations involved were in accord with international standards. There was undoubtedly human error involved somewhere in the fuelling function, but the precise details of this error will probably not be determined.

The actions taken since the accident provide assurance that this type of incident is unlikely to recur.

3. CONCLUSIONS

3.1 *Findings*

1. The aircraft was correctly maintained and so far as the operation of the aircraft at the time of the accident was concerned there was no relevant defect.

2. As refuelling commenced the starboard inboard refuelling hose swivel joint became detached and 72 U.S. gallons of fuel were spilled. The fuel ignited in the tailpipe of Nr 3 engine due to the temperature within the engine being above the spontaneous ignition temperature of the fuel and a large ground fire developed.

3. The swivel joint became disconnected probably because it was not in the fully locked position when the fuel delivery hose was connected to the aircraft. The coupling was secure enough to permit connection to the aircraft but when the fuel delivery commenced the coupling separated.

4. The subject refuelling operation was the first one following the weekly hose end strainer check and it is probable that on completion of the check the quick disconnect coupling was not properly locked. Due to the design of coupling which incorporates two guarded locking devices, it is unlikely that the coupling separated because of striking some object.

5. After the fire started an emergency evacuation was ordered by the Captain and the evidence is that it was carried out effectively. Emergency slides were

used at only doors 1 right and 5 left because smoke or vehicles precluded the use of the other doors. The crew's evidence is that hand baggage and duty free goods hampered the evacuation.

3.2 *Cause*

This accident was caused as the Swivel joint became disconnected probably because it was not in the locked position when the fuel delivery hose was connected to the aircraft. The coupling was secure enough to permit connections to the aircraft but when fuel delivery commenced the coupling separated. As the fuel hose fell away fuel ignited in the tailpipe of the Nr 3 engine due to the temperature within the engine being above the spontaneous ignition temperature of the fuel.

4. SAFETY RECOMMENDATIONS

4.1 Immediately the nature of the accident was known all stations were notified of the necessity to check the security of all quick disconnect couplings prior to every refuelling operation.

In February 1979, Shell International Trading Company advised all Shell Companies to carry out immediate modifications to all makes of quick disconnect couplings to increase the security. The modification to a Flight Refuelling coupling requires the fitting of a spring clip to stop the toggle lever being depressed, thus preventing the collar from being rotated. Details of this modification have been issued by Flight Refuelling to every purchaser of their couplings.

4.2 *Recommendations*

Because of the action taken immediately following the cause of the accident (see para 4.1) no recommendations in that respect have been made. It is however necessary to make the following:–

1. To avoid confusions and to warn the Flight Deck Crew of any problems, it is recommended that the Cockpit/Ground Contact (Comms) must be established and maintained throughout refuelling especially when passengers are on board.

2. Appropriate action must be taken to restrict the size and quantity of items of carry on luggage on aircraft.

ICAO Ref: 441/78

No. 2

McDonnell-Douglas DC-8-61, N8082U, accident at Portland, Oregon, United States, on 28 December 1978.
Report No. NTSB-AAR-79-7, dated 7 June 1979, released by the National Transportation Safety Board, United States.

SYNOPSIS

About 1815 Pacific standard time on December 28, 1978, United Airlines, Inc., Flight 173 crashed into a wooded, populated area of suburban Portland, Oregon, during an approach to the Portland International Airport. The aircraft had delayed southeast of the airport at a low altitude for about 1 hour while the flightcrew coped with a landing gear malfunction and prepared the passengers for the possibility of a landing gear failure upon landing. The plane crashed about 6 nmi southeast of the airport. The aircraft was destroyed; there was no fire. Of the 181 passengers and 8 crewmembers aboard, 8 passengers, the flight engineer, and a flight attendant were killed and 21 passengers and 2 crewmembers were injured seriously.

The National Transportation Safety Board determined that the probable cause of the accident was the failure of the captain to monitor properly the aircraft's fuel state and to properly respond to the low fuel state and the crewmember's advisories regarding fuel state. This resulted in fuel exhaustion to all engines. His inattention resulted from preoccupation with a landing gear malfunction and preparations for a possible landing emergency.

Contributing to the accident was the failure of the other two flight crewmembers either to fully comprehend the criticality of the fuel state or to successfully communicate their concern to the captain.

1. FACTUAL INFORMATION

1.1 History of the Flight

On December 28, 1978, United Airlines, Inc., Flight 173, a McDonnell-Douglas DC-8-61 (N8082U), was a scheduled flight from John F. Kennedy International Airport, New York, to Portland International Airport, Portland, Oregon, with an en route stop at Denver, Colorado.

Flight 173 departed from Denver about 1447[1/] with 189 persons on board, including 6 infants, and 8 crewmembers. The flight was cleared to Portland on an instrument flight rules (IFR) flight plan. The planned time en route was 2 hrs 26 min. The planned arrival time at Portland was 1713.

According to the automatic flight plan and monitoring system[2/] the total amount of fuel required for the flight to Portland was 31,900 lbs. There was 46,700 lbs of fuel on board the aircraft when it departed the gate at Denver. This fuel included the Federal Aviation Regulation requirement for fuel to destination plus 45 min and the company contingency fuel of about 20 min. During a postaccident interview, the captain stated that he was very close to his predicted fuel for the entire flight to Portland "... or there would have been some discussion of it." The captain also explained that his flight from Denver to Portland was normal.

At 1705:47, Flight 173 called Portland Approach and advised that its altitude was 10,000 ft 3/ and its airspeed was being reduced. Portland responded and told the flight to maintain its heading for a visual approach to runway 28. Flight 173 acknowledged the approach instructions and stated, ". . . we have the field in sight."

At 1707:55, Portland Approach instructed the flight to descend and maintain 8,000 ft. Flight 173 acknowledged the instructions and advised that it was "leaving ten." At 1709:40, Flight 173 received and acknowledged a clearance to continue its descent to 6,000 ft.

During a postaccident interview, the captain stated that, when Flight 173 was descending through about 8,000 ft, the first officer, who was flying the aircraft, requested the wing flaps be extended to 15°, then asked that the landing gear be lowered. The captain stated that he complied with both requests. However, he further stated that, as the landing gear extended, ". . . it was noticeably unusual and (I) feel it seemed to go down more rapidly. As (it is) my recollection, it was a thump, thump in sound and feel. I don't recall getting the red and transient gear door light. The thump was much out of the ordinary for this airplane. It was noticeably different and we got the nose gear green

1/ All times herein are Pacific standard, based on the 24-hour clock.
2/ A computer printout which predicted the amount of fuel that would be used between several identifiable en route points. The flightcrew was able to check the actual fuel used against the predicted fuel use at each of these points.
3/ All altitudes are mean sea level unless otherwise indicated.

light but no other lights." The captain also said the first officer remarked that the aircraft "yawed to the right. . . ." Flight attendant and passenger statements also indicate that there was a loud noise and a severe jolt when the landing gear was lowered.

At 1712:20, Portland Approach requested, "United one seven three heavy, contact the tower (Portland), one one eight point seven." The flight responded, "negative, we'll stay with you. We'll stay at five. We'll maintain about a hundred and seventy knots. We got a gear problem. We'll let you know." This was the first indication to anyone on the ground that Flight 173 had a problem. At 1712:28, Portland Approach replied, "United one seventy-three heavy roger, maintain five thousand. Turn left heading two zero zero." The flight acknowleged the instructions.

At 1714:43, Portland Approach advised, "United one seventy three heavy, turn left heading, one zero zero and I'll just orbit you out there 'til you get your problem." Flight 173 acknowledged the instructions.

For the next 23 min, while Portland Approach was vectoring the aircraft in a holding pattern south and east of the airport, the flightcrew discussed and accomplished all of the emergency and precautionary actions available to them to assure themselves that all landing gear was locked in the full down position. The flt. engineer checked the visual indicators on top of both wings, which extend above the wing surface when the landing gear is down-and-locked.

The captain stated that during this same time period, the first flight attendant came forward and he discussed the situation with her. He told her that after they ran a few more checks, he would let her know what he intended to do.

About 1738, Flight 173 contacted the United Airlines Systems Line Maintenance Control Center in San Francisco, California, through Aeronautical Radio, Inc. 4/ According to recordings, at 1740:47 the captain explained to company dispatch and maintenance personnel the landing gear problem and what the flightcrew had done to assure that the landing gear was fully extended. He reported about 7,000 lbs of fuel on board and stated his intention to hold for another 15 or 20 minutes. He stated that he was going to have the flight attendants prepare the passengers for emergency evacuation.

At 1744:03, United San Francisco asked, "okay, United one seventy three ... You estimate that you'll make a landing about five minutes past the hour. Is that okay?" The captain responded, "Ya, that's good ball park. I'm not gonna hurry the girls. We got about a hundred sixty five people on board and we . . .want to . . .take our time and get everybody ready and then we'll go. It's clear as a bell and no problem."

4/ Aeronautical Radio, Inc., an air-to-ground radio service which provides a communication system for commercial aircraft.

The aircraft continued to circle under the direction of Portland Approach in a triangular pattern southeast of the airport at 5,000 ft. The pattern kept that aircraft within about 20 nmi of the airport. (See Figure 1.)

From about 1744:30 until about 1745:23, the cockpit voice recorder (CVR) contained conversation between the captain and the first flight attendant concerning passenger preparation, crash landing procedures, and evacuation procedures. During his initial interview, the captain indicated that he neither designated a time limit to the flight attendant, nor asked her how long it would take to prepare the cabin. He stated that he assumed 10 or 15 minutes would be reasonable and that some preparations could be made on the final approach to the airport.

At 1746:52, the first officer asked the flight engineer, "How much fuel we got . . . ?" The flight engineer responded, "Five thousand." The first officer acknowledged the response.

At 1748:38, Portland Approach advised Flight 173 that there was another aircraft in its vicinity. The first officer advised Portland Approach that he had the aircraft in sight.

At 1748:54, the first officer asked the captain, ". . .what's the fuel show now . . . ?" The captain replied, "Five." The first officer repeated, "Five." At 1749, after a partially unintelligible comment by the flight engineer concerning fuel pump lights, the captain stated, "That's about right, the feed pumps are starting to blink." According to data received from the manufacturer, the total usable fuel remaining when the inboard feed pump lights illuminate is 5,000 lbs. At this time, according to flight data recorder (FDR) and air traffic control data, the aircraft was about 13 nmi south of the airport on a west southwesterly heading.

From just after 1749 until 1749:45, the flightcrew engaged in further conversation about the status of the landing gear. This conversation was interrupted by a heading change from Portland Approach and was followed by a traffic advisory from Portland Approach.

About 1750:20, the captain asked the flight engineer to "Give us a current card on weight. Figure about another fifteen minutes." The first officer responded, "Fifteen minutes?" To which the captain replied, "Yeah, give us three or four thousand pounds on top of zero fuel weight." The flight engineer then said, "Not enough. Fifteen minutes is gonna—really run us low on fuel here." At 1750:47, the flight engineer gave the following information for the landing data card: "...Okay. Take three thousands pounds, two hundred and four." At this time the aircraft was about 18 nmi south of the airport in a turn to the northeast.

At 1751:35, the captain instructed the flight engineer to contact the company representative at Portland and apprise him of the situation and tell him that Flight 173 would land with about 4,000 lbs of fuel. From

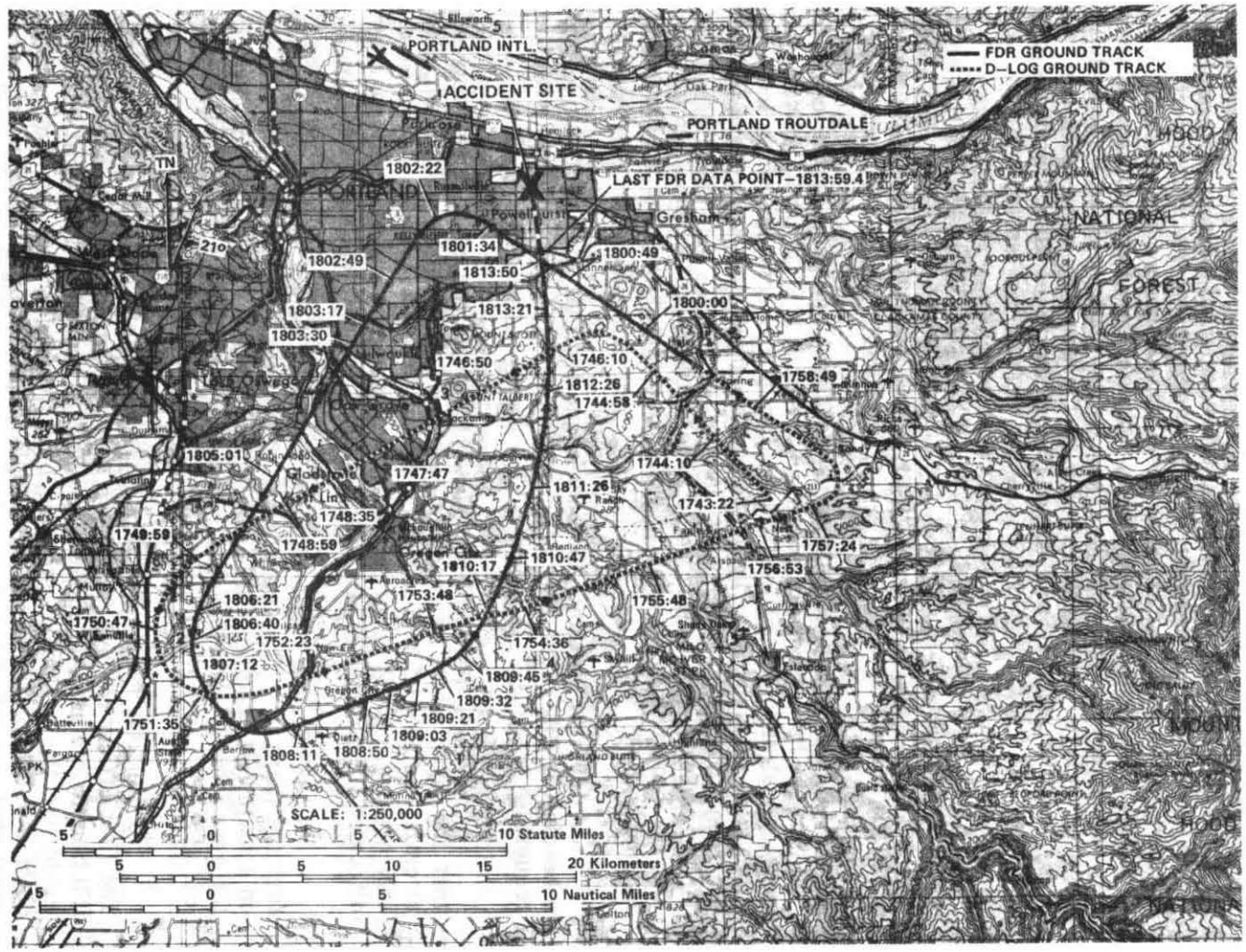


Figure 1. -- Flight track of N8082U.

1752:17 until about 1753:30, the flight engineer talked to Portland and discussed the aircraft's fuel state, the number of persons on board the aircraft, and the emergency landing preparations at the airport. At 1753:30, because of an inquiry from the company representative at Portland, the flight engineer told the captain, "He wants to know if we'll be landing about five after." The captain replied, "Yes." The flight engineer relayed the captain's reply to the company representative. At this time the aircraft was about 17 nmi south of the airport heading northeast.

At 1755:04, the flight engineer reported the "...approach descent check is complete." At 1756:53, the first officer asked, "How much fuel you got now?" The flight engineer responded that 4,000 lbs remained, 1,000 lbs in each tank.

At 1757:21, the captain sent the flight engineer to the cabin to "...kinda see how things are going. . . ." From 1757:30 until 1800:50, the captain and the first officer engaged in a conversation which included discussions of giving the flight attendants ample time to prepare for the emergency, cockpit procedures in the event of an evacuation after landing, whether the brakes would have antiskid protection after landing, and the procedures the captain would be using during the approach and landing.

At 1801:12, Portland Approach requested that the flight turn left to a heading of 195°. The first officer acknowledged and complied with the request.

At 1801:34, the flight engineer returned to the cockpit and reported that the cabin would be ready in "another two or three minutes." The aircraft was about 5 nmi southeast of the airport turning to a southwesterly heading. Until about 1802:10, the captain and the flight engineer discussed the passengers and their attitudes toward the emergency.

At 1802:22, the flight engineer advised, "We got about three on the fuel and that's it." The aircraft was then about 5 nmi south of the airport on a southwest heading. The captain responded, "Okay. On touchdown, if the gear folds or something really jumps the track, get those boost pumps off so that. . .you might even get the valves open."

At 1802:44, Portland Approach asked Flight 173 for a status report. The first officer replied, "Yeah, we have indication our gear is abnormal. It'll be our intention, in about five minutes, to land on two eight left. We would like the equipment standing by. Our indications are the gear is down and locked. We've got our people prepared for an evacuation in the event that should become necessary."

At 1803:14 Portland Approach asked that Flight 173 advise them when the approach would begin. The captain responded, ". . . They've about finished in the cabin. I'd guess about another three, four, five

minutes." At this time the aircraft was about 8 nmi south of the airport on a southwesterly heading.

At 1803:23, Portland Approach asked Flight 173 for the number of persons on board and the amount of fuel remaining. The captain replied, ". . . about four thousand, well, make it three thousand, pounds of fuel," and "you can add to that one-seventy-two plus six laps—infants."

From 1803:38 until 1806:10, the flightcrew engaged in a conversation which concerned (1) checking the landing gear warning horn as further evidence that the landing gear was fully down and locked and (2) whether automatic spoilers and antiskid would operate normally with the landing gear circuit breakers out.

At 1806:19, the first flight attendant entered the cockpit. The captain asked, "How you doing?" She responded, "Well, I think we're ready." At this time the aircraft was about 17 nmi south of the airport on a southwesterly heading. The conversation between the first flight attendant and the captain continued until about 1806:40 when the captain said, "Okay. We're going to go in now. We should be landing in about five minutes." Almost simultaneous with this comment, the first officer said, "I think you just lost number four . . . ," followed immediately by advice to the flight engineer, ". . . better get some crossfeeds open there or something."

At 1806:46, the first officer told the captain, "We're going to lose an engine. . . ." The captain replied, "Why?" At 1806:49, the first officer again stated, "We're losing an engine." Again the captain asked, "Why?" The first officer responded, "Fuel."

Between 1806:52 and 1807:06, the CVR revealed conflicting and confusing conversation between flight crewmembers as to the aircraft's fuel state. At 1807:06, the first officer said, "It's flamed out."

At 1807:12, the captain called Portland Approach and requested, ". . .would like clearance for an approach into two eight left, now." The aircraft was about 19 nmi south southwest of the airport and turning left. This was the first request for an approach clearance from Flight 173 since the landing gear problem began. Portland Approach immediately gave the flight vectors for a visual approach to runway 28L. The flight turned toward the vector heading of 010°.

From 1807:27 until 1809:16, the following intracockpit conversation took place:

1807:27 - Flight Engineer: "We're going to lose number three in a minute, too."
1807:31 - Flight Engineer: "It's showing zero."
Captain: "You got a thousand pounds. You got to."

Flight Engineer: "Five thousand in there . . .but we lost it."
Captain: "Alright."
1807:38 - Flight Engineer: "Are you getting it back?"
1807:40 - First Officer: "No number four. You got that crossfeed open?"
1807:41 - Flight Engineer: "No, I haven't got it open. Which one?"
1807:42 - Captain: "Open 'em both--get some fuel in there. Got some fuel pressure?"
Flight Engineer: "Yes, sir."
1807:48 - Captain: "Rotation. Now she's coming."
1807:52 - Captain: "Okay, watch one and two. We're showing down to zero or a thousand."
Flight Engineer: "Yeah"
Captain: "On number one?"
Flight Engineer: "Right."
1808:08 - First Officer: "Still not getting it."
1808:11 - Captain: "Well, open all four crossfeeds."
Flight Engineer: "All four?"
Captain: "Yeah."
1808:14 - First Officer: "Alright, now it's coming."
1808:19 - First Officer: "It's going to be --on approach though."
Unknown Voice: "Yeah."
1808:42 - Captain: "You gotta keep 'em running. . . ."
Flight Engineer: "Yes, sir."
1808:45 - First Officer: "Get this . . .on the ground."
Flight Engineer: "Yeah. It's showing not very much more fuel."
1809:16 - Flight Engineer: "We're down to one on the totalizer. Number two is empty."

At 1809:21, the captain advised Portland Approach, "United, seven three is going to turn toward the airport and come on in." After confirming Flight 173's intentions, Portland Approach cleared the flight for the visual approach to runway 28L.

At 1810:17, the captain requested that the flight engineer "reset that circuit breaker momentarily. See if we get gear lights." The flight engineer complied with the request.

At 1810:47, the captain requested the flight's distance from the airport. Portland approach responded, "I'd call it eighteen flying miles." At 1812:42, the captain made another request for distance. Portland Approach responded, "Twelve flying miles." The flight was then cleared to contact Portland tower.

At 1813:21, the flight engineer stated, "We've lost two engines, guys." At 1813:25, he stated, "We just lost two engines - one and two."

At 1813:38, the captain said, "They're all going. We can't make Troutdale." 5/ The first officer said, "We can't make anything."

5/ A small airport on the final approach path to runway 28L.

At 1813:46, the captain told the first officer, "Okay. Declare a mayday." At 1813:50, the first officer called Portland International Airport tower and declared, "Portland tower, United one seventy three heavy, Mayday. We're--the engines are flaming out. We're going down. We're not going to be able to make the airport." This was the last radio transmission from Flight 173.

About 1815, the aircraft crashed into a wooded section of a populated area of suburban Portland about 6 nmi east southeast of the airport. There was no fire. The wreckage path was about 1,554 ft long and about 130 ft wide.

The accident occurred during the hours of darkness at latitude 45°31'21"N and longitude 122°29'59"W. The elevation of the accident site was 285 ft.

1.2 Injuries to Persons

Injuries	Crew	Passengers	Others
Fatal	2	8	0
Serious	2	21	0
Minor/None	4	152	0

1.3 Damage to Aircraft

The aircraft was destroyed.

1.4 Other Damage

Two unoccupied homes were destroyed. Telephone lines were cut and high-tension electrical powerlines were damaged.

1.5 Personnel Information

The crewmembers were properly certificated and qualified for the flight.

1.6 Aircraft Information

The aircraft was certificated and maintained in accordance with Federal Aviation Administration (FAA) requirements. The gross weight and center of gravity were within prescribed limits for the approach and landing. There was no usable fuel in the aircraft when it crashed.

The figures below illustrate the aircraft's approximate takeoff gross weight, approximate landing weight, and the approximate pounds of fuel remaining upon arrival in the vicinity of the Portland International Airport; these figures are based on normal operations.

	lbs
Zero fuel weight from weight manifest	201,927
Total fuel on board from fuel service form	+46,700
Aircraft weight before departure from gate at Denver	248,627
Fuel consumption on taxi	- 1,000
Takeoff gross weight	247,627
Fuel consumption en route to Portland, based on flight plan	-31,900
Landing weight at Portland	215,727
Zero fuel weight from weight manifest	-201,927
Fuel remaining at Portland	13,800

Throughout the landing delay, Flight 173 remained at 5,000 ft with landing gear down and flaps set at 15°. Under these conditions, the Safety Board estimated that the flight would have been burning fuel at the rate of about 13,209 lbs per hour--220 lbs per min. At the beginning of the landing delay, there were about 13,334 lbs of fuel on board.

A new type of fuel quantity indicating system was retrofitted to this aircraft on May 12, 1978. The retrofit was authorized by Change Order Authorization No. 2-4849. With the new system installed, there are eight individual tank quantity gages. Each of these gages has three digits which are seven-segment incandescent lamps. On these individual tank gages, the digital reading is multiplied by 100 to obtain the total amount of fuel in the tank.

The totalizer gage receives input from each individual tank gage and displays the total fuel available on three digital readouts. However, this digital reading must be multiplied by 1,000 to obtain the value of the total amount of fuel on board. The smallest increment of change that can be indicated on the individual tank gages is 100 lbs. The smallest increment of change on the totalizer is 1,000 lbs.

Before the implementation of the change order, each individual tank gage displayed five digits which were read directly to obtain the amount of fuel in each tank.

The change order also replaced the flight engineer's totalizer gage, which had displayed six digits read directly for total fuel on board.

According to United Airlines, the primary purposes of installing the fuel quantity indicating system were (1) to reduce erroneous system indications because of stray pickup of 400 Hz signals in the fuel quantity indicating system wiring, and (2) to reduce indication errors from current leaks across the elements of the capacitive probes and compensators.

After the accident, United Airlines determined that the aircraft was burning fuel in accordance with the automatic flight plan and monitoring system. In October 1978, fuel burnoff examination indicated that the aircraft was not consuming fuel as fast as predicted; it was 1.04 percent

less than predicted. In addition, another method for determining burnout was begun by United engineers. Each trip's total burnout was divided by total time. For December 1978 these aggregate values verified that this aircraft's fuel consumption was within 1 percent of the plan.

The aircraft was also equipped with a fuel flow indicator for each engine which displayed, in hundreds of pounds, the hourly rate of fuel being used by the engine. These indicators were located on the pilot's forward engine instrument panel along with other engine monitoring gages.

1.7 Meteorological Information

Surface weather observations taken before and after the accident at Portland International Airport by National Weather Service personnel were:

1655 - 4,500 ft scattered; visibility — 30 mi; temperature — 30°F; dewpoint — 13°F; wind — 340° at 8 kns; altimeter setting — 30.16 inHg.

1755 - Clear; visibility — 15 mi; temperature — 29°F; dewpoint--13°F; wind — 010° at 11 kns; altimeter setting — 30.17 inHg.

1829 Local - Clear; visibility — 15 mi; temperature — 28°F; dewpoint--12°F; wind — 350° at 11 kns; altimeter setting — 30.19 inHg; AIRCRAFT MISHAP.

1.8 Aids to Navigation

During his deposition, the captain stated that he had set the Portland VORTAC, which is located 9.2 nmi north-northeast of Portland International Airport, in both of his VOR receivers. He stated also that he was receiving distance measuring equipment information.

1.9 Communications

No communications difficulties were reported.

1.10 Aerodrome Information

Runway 28L at Portland International Airport is hard surfaced and is 11, 014 ft long and 150 ft wide. The published touchdown zone elevation and field elevation are 19 ft and 26 ft, respectively. The runway is equipped with high intensity runway edge lights, centerline lights, and visual approach slope indicator lights. The airport has two other runways. Runway 10L/28R, which is parallel to runway 28L, is 8,004 ft long and 150 ft wide. It is the primary instrument runway. Runway 02/20 is 7,000 ft long and 150 ft wide. It is used mainly as a crosswind runway.

The airport is located near the south shore of the Columbia River north of Portland. The terrain southeast of the airport is characterized by low rolling hills, which rise from the river valley.

1.11 Flight Recorders

N8082U was equipped with a Fairchild model 5424 flight data recorder (FDR), serial No. 6043. The recorder showed no outward evidence of damage. The foil recording medium was not damaged; all parameter and binary traces were present and active with no evidence of recorder malfunction or recording abnormalities. Electrical power to the recorder was terminated about 44 sec before the aircraft crashed. A readout was made of the final 15 min 44.7 sec of the recorded traces. This readout covers the 15 min of flight before all parameter traces — altitude, airspeed, magnetic heading, and vertical acceleration — ceased to be recorded and continues for an additional 44.7 sec where all binary traces became atypical.

N8082U was also equipped with a Sundstrand model V557 CVR, serial No. 1427. The recorder was removed from the aircraft and the entire tape was transcribed. The quality of the recording was good. (See Appendix D.)

1.12 Wreckage and Impact Information

The aircraft first struck two trees about 100 ft above the ground. These trees were about 1,554 ft from the point where the wreckage came to rest. About 541 ft farther along the flightpath on a heading of about 345°, the aircraft struck two trees about 85 ft above the ground. About 400 ft farther, the right wing struck a tree about 45 ft above the ground. About 225 ft beyond that point, the left outer wing struck a tree about 8 ft above the ground. The aircraft then struck and destroyed an unoccupied house which was located about 1,230 ft from the first tree. Pieces of the aircraft's left wing structure were located just beyond the house.

The two main landing gear and the nose section of the aircraft first struck a 5-ft embankment next to a city street about 1,275 ft from the first tree. The aircraft continued across the street and came to rest on a heading of 330° between some trees and on top of another unoccupied house. The tail of the aircraft came to rest about 1,350 ft from the first tree. Just after crossing the street, the verticle stabilizer struck a series of high tension cables, which ran parallel to the street.

The fuselage, from about the fifth row of passenger seats forward, sustained severe, extensive impact damage in a generally rearward direction. The cockpit upper structure, which included the cockpit forward windows, had separated and was found to the right of the fuselage just forward of the inboard end of the right wing. The cockpit floor structure, which included portions of the crew seats, sections of the instrument panel, and the nose tunnel structure with the nose gear assembly partially attached, had separated and rotated to the right and aft. This structure was in a partially inverted position. All portions of the fuselage structure were accounted for and all of the structural damage was caused by impact with the ground and the numerous large trees in the immediate area.

The lower left side of the fuselage, between the fourth and sixth rows of passenger seats and below window level, had been torn away. The remainder of the underside of the fuselage sustained heavy damage from contact with several large trees and tree stumps. The passenger cabin interior, from row 6 to the aft bulkhead, was relatively intact. At several points along the fuselage, windows were smashed and the fuselage had been dented by large trees and separated portions of the main landing gear.

The empennage showed moderate impact damage. The vertical stabilizer leading edge had been damaged by high tension cables at three points just forward of the upper three rudder-to-stabilizer hinge points.

The left wing had separated from the fuselage about 3 ft outboard from the fuselage attachment point. The No. 2 engine had separated from its pylon and was located adjacent to the wing trailing edge. The No. 1 engine remained attached to a section of left wing structure. A 7-ft-long section of the left wingtip had been sheared off and was found near the first house.

The right wing separated about 5 ft from the fuselage. A 2-ft opening was evident between the fuselage and wing leading edge structure. The wing leading edge, from a point about 5 ft outboard from the leading edge inboard end, was cut and torn aft to the front spar assembly. A large section of right wing leading edge structure had separated during the impact sequence and was also found near the first house.

A section of right wing with the No. 3 engine and pylon attached was located just forward of the right horizontal stabilizer. The outboard wing section, which included the No. 4 engine, was to the right of the fuselage.

All four engines were inspected and found to be capable of operation. None showed signs of rotation at impact.

Both main landing gear were fully extended but were torn from their mounting structures. They were located near main wreckage. Inspection of the right main landing gear retraction mechanism showed corrosion in the threads of an attachment eyebolt. The eyebolt was pulled out of the actuator cylinder piston. The nose landing gear was fully extended and remained attached to the nose tunnel structure.

1.13 Medical and Pathological Information

A review of the flightcrew's medical records revealed no evidence of medical problems that might have affected their performance.

The 10 persons who were killed in the crash died from impact trauma. Toxicological analyses showed no acidic, neutral, or basic drugs or ethanol in the blood taken from the flight engineer and first flight attendant.

1.14 Fire

There was no fire.

1.15 Survival Aspects

The accident was partially survivable. The 10 occupants killed in the crash were located between the flight engineer's station in the cockpit and row 5 in the passenger cabin. All of the passengers who were killed had been located on the right side of the cabin. That section of the aircraft was destroyed during the accident sequence.

The most seriously injured passengers were seated in the right forward portion of the cabin near an area of the fuselage which appeared to have been penetrated by a large tree. These persons were seated near those passengers who were injured fatally. Some seriously injured passengers were seated in the rear cabin near the trailing edge of the wings. The fuselage in this area had been penetrated and the floor and seats had been disrupted.

Some passengers sustained serious injuries during the evacuation. Two passengers sustained fractures and others sustained lacerations and abrasions when they either fell from exits or as they climbed through debris outside the aircraft in order to reach the ground. As a result of the accident, 22 persons were admitted to hospitals with serious injuries ranging from multiple fractures of extremities and fractures of cervical vertebrae, to observations for possible injuries.

The plane crashed in the jurisdiction of Multnomah County Rural Fire Protection District No. 10. Three fire departments sent personnel and equipment to the scene: The Port of Portland (Airport) Fire Department; Multnomah RFPD No. 10, and the City of Portland Fire Bureau. A total of 39 fire units and 108 on-duty fire personnel responded to the scene. Numerous off-duty fire personnel from all fire departments also responded to the scene. Because there was no fire, the basic fire service functions were search and rescue, extrication, triage, emergency medical care, precautionary foaming of some aircraft parts and surrounding area, laying standby firefighting water supply lines, transporting or assisting ambulatory victims to a nearby church, setting up area lighting, providing some interagency radio communications, and setting up the on-scene command post.

Although there were many occupied houses and apartment complexes in the immediate vicinity of the accident, there were no ground casualties and no postcrash fire. Injured persons were transported to nearby hospitals by helicopter and ambulance.

The aircraft was equipped with 10 floor level exits, each provided with automatically inflatable emergency escape slides. In addition to slides at the boarding doors (1L and 5L) and at the two galley service

doors (2R and 5R), slides were located at the six "jet escape" floor level exits (1R, 2L, 3L, 3R, 4L, 4R). The "jet escape" doors were hinged at the bottom and were designed to swing down and outward when opened. Movement of the door actuated the automatic inflation of the escape slide.

The slide from exit 1L was found wrapped around a tree at the left wing. The slides from exits 3L and 3R were found packed and uninflated. These exits were reportedly blocked by debris outside the aircraft. The slide at exit 1R reportedly inflated inside the cabin and extended across the aisle and lodged against seat 8C. The door was prevented from opening fully because of cockpit and forward cabin debris outside the aircraft. The slide at exit 2R also reportedly inflated inside the cabin and blocked the cabin aisle. The exit door was displaced inward when the plane hit a tree.

The slide at exit 4R reportedly inflated inside the cabin when the door was opened by a passenger. The slide inflated upward and partially blocked the exit opening. Because of debris outside the fuselage, the exit door was prevented from opening fully. The passenger who opened the door reported that about 10 persons used this exit before the slide was pushed out the exit and onto debris. The remaining escape slides reportedly deployed outside the aircraft and inflated but were punctured or torn by debris during the evacuation.

The escape slides were removed from the accident site and were examined on January 3, and on January 9, 1979. No discrepancies were found in the installation, maintenance, manufacture, or design of the escape slides.

The evacuation was completed in about 2 min. Except for seats at rows 20 through 22 which were torn loose from the floor attachments, there was only minor disruption of the cabin furnishings aft of row 6. The emergency lights provided adequate illumination during the evacuation.

1.16 Tests and Research

1.16.1 Retract Cylinder Assembly

The Safety Board examined the piston rod and the mating end from the right main landing gear retract cylinder assembly at its metallurgical laboratory in Washington, D.C. The examination showed that the primary cause of the separation of the rod end from the piston rod was severe corrosion caused by moisture on to the mating threads of both components. As a result of the corrosion, the joint was weakened to such an extent that only a comparatively low tensile load was required to pull the rod end out of the piston rod. The pattern of mechanical damage indicated that all of the rod threads had been engaged and that the rod end had been pulled straight out of the piston rod without any significant rotation.

1.16.2 Fuel Control Test

Functional testing of the fuel controllers from each of the four engines was conducted at the company's maintenance base. No discrepancies or out-of-tolerance conditions were found.

1.16.3 Aircraft Systems Examinations

During the week of January 8, 1979, the following examinations were conducted at the company's maintenance facility:

(1) Fuel Flow Indicators

The shop examinations confirmed that the four indicators were indicating zero fuel flow. The front face, case, and electrical connections were all damaged and none could be operated or tested before being repaired. After minor repairs to the electrical connections only the flow meter for the No. 2 engine became operable, and it met the linearity specifications.

(2) Fuel Quantity Gages

During the wreckage salvage, the eight fuel quantity gages were recovered. All units were damaged and repairs were attempted on each. Three gages could be repaired sufficiently to allow functional testing.

The No. 4 main tank gage was given a lamp check, segment check, and self-test check and all were within specification. In addition, a linearity check was made at full, at 1,000 lbs, at 500 lbs, and at empty. The results were within specifications. The No. 4 alternate tank gage and the No. 2 main tank gage were tested in the same manner, and the results were within specifications. The other tank gages were not operable because of damage and, therefore, could not be tested.

(3) Totalizer Gage

The glass face was broken, the electrical connector bent, and the case punctured. The damage was too extensive to enable repair for testing.

(4) No. 1 Main Fuel Tank Capacitance Probes

The five capacitance probes from the No. 1 main fuel tank were examined according to company specification. All units were within specification except probe No. MR 28062, serial No. 525856-31X. This unit did not meet resistance tolerance specification when wet.

(5) Tank Reference Capacitors

One of these units is located in each main tank. All four units were recovered and, when tested, met specifications.

(6) Right Main Landing Gear Down-Lock Switch

This switch is activated when the gear reaches a down-and-locked position. A similar switch was installed on the left main landing gear. A comparison of the damage to the two switch cases showed that the switch from the right landing gear had been damaged apparently by an internal part that pushed the case outward and had distorted it. Electrical tests of the switch and attached wiring indicated an intermittent short circuit when the switch was shaken. X-rays of the switch showed that a large spring had become free of its mounts within the switch case. Normally this spring returns the down-lock switch to the gear-not- down position when the landing gear is retracted. The switch case was cut open and several coils of the spring were found spread apart. When the spring and switch case damages were matched, one end of the spring fit into the distorted portion of the case. The other end of the spring touched wiring terminal No. 8 of the microswitch and marks indicative of electrical arcing were found on the spring where it contacted terminal No. 8.

(7) Left Main Landing Gear Down-Lock Switch

The spring of the left main landing gear down lock switch was free of its mounts. The coils of this spring were not bent and no marks similar to electrical arcing were found.

(8) Landing Gear Warning and Interlock Circuit Breaker

When examined in the field, this circuit breaker appeared to be mechanically extended or electrically open. There was some mechanical damage. Later, shop tests verified that the circuit breaker was open. It could be operated mechanically and it opened and closed the electrical circuit properly.

(9) Distance Measuring Equipment

Two distance measuring equipment units, Collins Model 860E-2, serial No. 3954 (No. 1) and serial No. 617 (No. 2), were opened in the company maintenance shops and the distance modules were removed. When connected to a test panel, the mileage readouts were 16.05 nmi for the No. 1 unit and 16.0 nmi for the No. 2 unit. Both units were selected to channel 113X, which corresponds to a VOR frequency of 116.60 MHz.

1.16.4 Fuel Quantity System Error

Upon request, United Airlines provided the Safety Board with an error analysis of the fuel quantity indicating system for the accident aircraft. Analyses were prepared for three different assumptions. The

first analysis assumed that all errors were at their limits and in the same direction. The second analysis assumed that all errors were at their limits but were distributed randomly with respect to sign (root-sum-square analysis). The third analysis was a probable error analysis. All errors in this analysis were those associated with empty or near empty tanks.

These analyses indicated the following:

	Sum of Indicators		Totalizer	
Analysis Method	High Error lbs.	Low Error lbs.	High Error lbs.	Low Error lbs.
Worst-Case Error	2,283 High	1,482 Low	3,961 High	3,606 Low
Root-Square-Sum Error	828 High	28 Low	1,312 High	957 Low
Probable Error	685 High	185 High	1,239 High	885 Low

1.16.5 Fuel Burn Time History

At the request of the Safety Board, Douglas Aircraft Company and United Airlines studied fuel burn performance for the accident flight. In both studies, the fuel on board at the gate in Denver was 46,700 lbs. The fuel remaining at cruise at 35,000 ft was almost identical in both studies. United's calculations of fuel burn rate for the descent from 35,000 ft to the 5,000-ft holding altitude were 13 percent lower than Douglas'. However, United's fuel burn rate while in the holding pattern was 14 percent higher than Douglas'. This disparity was a result of different interpretations of meteorological and FDR data which resulted in differing mach values. Both studies had similar fuel remaining values when both flight recorders ceased operation; Douglas had calculated 178 lbs and United had calculated 73 lbs. Both studies compared favorably to the fuel burn time history computed by the Safety Board using information from the automatic flight plan and monitoring system and CVR data.

A correlation of CVR information with both fuel burn studies shows the observed and calculated fuel remaining values to be in agreement. The CVR transcript indicated an observed fuel remaining value of 5,000 lbs about 1749. The Douglas figure for that time was 5,250 lbs and United's was about 6,000 lbs. If the totalizer accumulated probable error of 885 lbs was applied, the calculated and observed fuel remaining values would be in agreement. In addition, the two studies indicated that the accident aircraft's fuel consumption was normal during the accident flight.

Although both studies had similar fuel remaining values when the aircraft lost its engine power, the Safety Board believes that the Douglas' study more closely approximates the fuel burn during the 5,000-ft hold period. Therefore, fuel remaining computations for this period are predicated on the manufacturer's figures of a calculated fuel burn of 13,209 lbs per hour (220 lbs per min). According to the manufacturer's study, the aircraft entered into the hold with about 13,334 lbs of fuel.

1.17 Other Information

1.17.1 Responsibility of the Crew

Excerpt from United Airlines Flight Operations Manual, paragraph 6.2, June 30, 1978:

"16. Except as otherwise specifically directed by the captain, all crew members noting a departure from prescribed procedures and safe practices should immediately advise the captain so that he is aware of and understands the particular situation and may take appropriate action."

1.17.2 United Airlines Flight Operations Bulletin 22-76, Fuel Policy Domestic FAR 45 Minute and Overwater/International 30 Minute Reserve Fuel.

"FAR 121.639 (C) does not specify in detail how the aforementioned requirements are to be calculated. United Airlines has established the following criteria for computing required fuel.

a. Weight - The operating weight empty of the airplane plus maximum structural payload or maximum space payload, whichever is smaller, plus the weight of the 45 minutes of reserve fuel.

b. Speed - Long range cruise speed.

c. Altitude - 25,000 Feet.

d. The ability to loiter at 5,000 feet at clean holding speed for 45 minutes."

"From the aforementioned criteria is derived the following DC-8-61 fuel requirements.

1. Fuel required for 45 minutes cruising at long range cruise at 25,000 feet is 8,300 pounds.
2. Fuel required for 45 minutes holding clean at 5,000 feet is 7,800 pounds.
3. FAR 45 minute reserve: 8,400 pounds."

1.17.3 Excerpts From United Airlines DC-8 Flight Manual

"Landing Gear Lever Down and Gear Unsafe Light On

If the visual down-lock indicators indicate the gear is down then a landing can be made at the captain's discretion." (Dated January 1, 1974, pg. I-44.)

"Landing Gear Apparently But Not Conclusively Down

If possible, have tower visually check. If there is reasonable indication that the gear is down then the landing can be made assuming gear is down. Do not taxi the airplane until gear locks have been installed." (Dated January 2, 1974, pg I-44-59.)

"Preparation For Evacuation

1. Notify ground station of emergency.

2. Advise the First Flight Attendant as to:

   a. nature of emergency and expected landing conditions,
   b. time available for preparation,
   c. signal for taking protective position,
   d. signal to be used if evacuation is not necessary,
   e. other special instructions.

3. Determine from the First Flight Attendant:

   a. the passenger load,
   b. number of infants, invalids, and other passengers who would be given special consideration.

4. Direct all nonessential cockpit members to move to the cabin and assist Flight Attendants as requested.

5. Review the EVACUATION Emergency procedure.

6. Make an announcement to the passengers as appropriate.

7. Accomplish the CABIN INSPECTION CHECKLIST below when advised by the First Flight Attendant that cabin preparations are completed.

8. Depressurize the airplane when below 10,000 feet.

9. Insure that the emergency exit lights switch is in the armed position.

10. Avoid landing, if possible, until emergency equipment and crews are standing by.

11. Advise the First Flight Attendant when approximately five minutes from landing.

12. Advise the passengers and Flight Attendants when to assume the protective position.

13. If evacuation is necessary, accomplish the EVACUATION Emergency Procedure."
(Dated September 27, 1975, page I-19.)

"Cabin Inspection Checklist

1. All Flight Attendants briefed on station, duties, and signals.

2. Passengers reseated as required and seats to be used by crew vacated.

3. Helper passengers briefed on station, duties, and exit operation.

4. Passengers briefed on:

   a. Protective position and signal to assume position.
   b. Seat belts tight and low.
   c. How to unfasten seat belts.
   d. Assigned exits and when and how to leave the airplane.

5. Passengers' glasses, dentures, high heels, and other possible hazardous items removed and stowed. Loose objects stowed in secure stowage areas.

6. Internal doors and curtains secured open.

7. Meal service furnishings in appropriate secure area.

8. Seat backs upright and tables stowed.

9. Pillows and blankets distributed for impact protection."
(Dated September 27, 1975, page I-20)

1.17.4 Excerpts From United Airlines Maintenance/Overhaul Manual

"Fuel Quantity Indicator System - Tolerance

a. All tanks at empty, ± 150 pounds.
b. Tank at full #1 & #4 Main ± 400 pounds
   #1 & #4 Alt ± 225 pounds
   #2 & #3 Main ± 400 pounds
   #2 & #3 Alt ± 250 pounds"
(Dated January 19, 1976, page 201.)

1.17.5 Main Landing Gear Retract Cylinder Assembly

Although the purpose of the main landing gear retract cylinder assembly is to raise the landing gear during the retract cycle, the hydraulic action of the cylinder acts as a buffer during the extend cycle to moderate the rate of extension and prohibit the landing gear from free falling to the down-and-locked position.

McDonnell-Douglas Corporation issued an All-Operator Letter, AOL 8-141, in July 1967, Main Landing Gear Retract Cylinder Assemblies, DC-8 Aircraft. The letter advised all DC-8 operators that several cylinder end fittings had been found with fractures in the thread roots. To alleviate this condition, the eyebolt threads were changed from machined to rolled-type threads. The letter also recommended sealing the threads with a corrosion resistant compound.

On March 27, 1968, McDonnell-Douglas issued Service Bulletin No. 32-131, DC-8 SC 1681, Landing Gear Extension and Retraction -Replace Main Landing Gear Retract Rod Assemblies. This bulletin provided information on the replacement of the retract cylinder rod end assemblies with machined threads with rod end assemblies with rolled-type threads.

In 1973, United Airlines instituted a gamma ray inspection program for the main landing gear retract actuating cylinder and rod ends on the DC-8 aircraft. The purpose of inspection was to detect thread corrosion in the cylinder. The cylinder threads on the main landing gear retract actuators of the accident aircraft were last inspected using the gamma ray inspection on April 2, 1977.

In order to provide additional threads and a longer eyebolt on actuator cylinders found with corrosion damage, the retract cylinder was to have been modified as provided for in the United Airlines Maintenance Manual dated January 2, 1974. The right main landing gear retract actuator on the accident aircraft had not been modified.

1.17.6 Dispatcher Responsibility and Authority

Under the provisions of United Airlines Flight Operations Manual, the flight dispatcher responsibility after the aircraft is airborne is limited to computation of fuel estimate under only two conditons—when contact is not established within 20 min and during a hijacking.

1.18 New Investigation Techniques

None

2. ANALYSIS

The flightcrew was properly certificated and each crewmember had received the training and the off-duty time prescribed by applicable regulations. There was no evidence of medical problems that might have affected their performance.

The aircraft was certificated and maintained according to applicable regulations. The gross weight and c.g. were within prescribed limits. Except for the failure of the piston rod on the right main landing gear retract cylinder assembly and the failure of the landing gear position indicating system, the aircraft's airframe, systems, structures, and powerplants were not factors in this accident.

The investigation revealed that fuel was burned at a normal rate between Denver and Portland. The aircraft arrived in the Portland area with the preplanned 13,800 lbs of fuel and began its delay at 5,000 ft with about 13,334 lbs.

The first problem which faced the captain of Flight 173 was the unsafe landing gear indication during the initial approach to Portland International Airport. This unsafe indication followed a loud thump, an abnormal vibration, and an abnormal aircraft yaw as the landing gear was lowered. The Safety Board's investigation revealed that the landing gear problem was caused by severe corrosion in the mating threads where the right main landing gear retract cylinder assembly actuator piston rod was connected to the rod end. The corrosion allowed the two parts to pull apart and the right main landing gear to fall free when the flightcrew lowered the landing gear. This rapid fall disabled the microswitch for the right main landing gear which completes an electrical circuit to the gear-position indicators in the cockpit. The difference between the time it took for the right main landing gear to free fall and the time it took for the the left main landing gear to extend normally, probably created a difference in aerodynamic drag for a short time. This difference in drag produced a transient yaw as the landing gear dropped.

Although the landing gear malfunction precipitated a series of events which culminated in the accident, the established company procedures for dealing with landing gear system failure(s) on the DC-8-61 are adequate to permit the safest possible operation and landing of the aircraft. Training procedures, including ground school, flight training, and proficiency and recurrent training, direct the flightcrew to the Irregular Procedures section of the DC-8 Flight Manual, which must be in the possession of crewmembers while in flight. The Irregular Procedures section instructed the crew to determine the position of both the main and nose landing gear visual indicators. "If the visual indicators indicate the gear is down, then a landing can be made at the captain's discretion." The flight engineer's check of the visual indicators for both main landing gear showed that they were down and locked. A visual check of the nose landing gear could not be made because the light which would have illuminated that down-and-locked visual indicator was not operating. However, unlike the main landing gear cockpit indicators, the cockpit indicator for the nose gear gave the proper "green gear-down" indication.

Admittedly, the abnormal gear extension was cause for concern and a flightcrew should assess the situation before communicating with the dispatch or maintenance personnel. However, aside from the crew's discussing the problem and adhering to the DC-8 Flight Manual, the only remaining step was to contact company dispatch and line maintenance. From the time the captain informed Portland Approach of the gear problem until contact with company dispatch and line maintenance, about 28 min had elapsed. The irregular gear check procedures contained in their manual were brief, the weather was good, the area was void of heavy traffic, and there were no additional problems experienced by the flight that would have delayed the captain's communicating with the company. The company maintenance staff verified that everything possible had been done to assure the integrity of the landing gear. Therefore, upon termination of communications with company dispatch and maintenance personnel, which was about 30 min before the crash, the captain could have made a landing attempt. The Safety Board believes that Flight 173 could have landed safely within 30 to 40 min after the landing gear malfunction.

Upon completing communications with company line maintenance and dispatch, the captain called the first flight attendant to the cockpit to instruct her to prepare the cabin for a possible abnormal landing. During the ensuing discussion, the captain did not assign the first flight attendant a specified time within which to prepare the cabin, as required by the flight manual. In the absence of such time constraint, the first flight attendant was probably left with the impression that time efficiency was not necessarily as important as the assurance of thorough preparation.

The Safety Board believes that any time a flight deviates from a flight plan, the flightcrew should evaluate the potential effect of such deviation on the aircraft fuel status. This flightcrew knew that the evaluation of the landing gear problem and preparation for an emergency landing would require extended holding before landing.

The flightcrew should have been aware that there were 46,700 lbs of fuel aboard the aircraft when it left Denver at 1433 and that there was about 45,650 lbs at takeoff at 1447. Regardless of whether they were aware of the actual fuel quantities, they certainly should have been aware that the initial fuel load was predicated on fuel consumption for the planned 2 hr 26 min en route flight, plus a reserve which includes sufficient fuel for 45 min at normal cruise and a contingency margin of about 20 min additional flight.

Therefore, the crew should have known and should have been concerned that fuel could become critical after holding. Proper crew management includes constant awareness of fuel remaining as it relates to time. In fact, the Safety Board believes that proper planning would provide for enough fuel on landing for a go-around should it become necessary. Such planning should also consider possible fuel-quantity indication inaccuracies. This would necessitate establishing a deadline time for initiating the approach and constant monitoring of time, as well as the aircraft's position relative to the active runway. Such procedures

should be routine for all flightcrews. However, based on available evidence, this flightcrew did not adhere to such procedures. On the contrary, the cockpit conversation indicates insufficient attention and a lack of awareness on the part of the captain about the aircraft's fuel state after entering and even after a prolonged period of holding. The other two flight crewmembers, although they made several comments regarding the aircraft's fuel state, did not express direct concern regarding the amount of time remaining to total fuel exhaustion. While there is evidence to indicate that the crew was aware of the amount of fuel remaining at various times, there is no evidence that the onboard quantity was monitored in relation to time remaining during the final 30 min of flight. The Safety Board believes that had the flightcrew been aware of the fuel state, comments concerning time to fuel exhaustion would have been voiced. However, there was none until after the aircraft was already in a position from which recovery was not possible.

In analyzing the flightcrew's actions, the Safety Board considered that the crew could have been misled by inaccuracies within the fuel-quantity measuring system. However, those intracockpit comments and radio transmissions in which fuel quantity was mentioned indicate that the fuel-quantity indicating system was accurate.

Had the flightcrew related any of these fuel quantities to fuel flow, they should have been aware that fuel exhaustion would occur at or about 1815. Other evidence that the captain had failed to assess the effect of continued holding on fuel state was provided by his stated intentions to land about 1805 with 4,000 lbs of fuel on board. Just minutes earlier, at 1748:56, he was made aware that only 5,000 lbs remained. During the 16 min between the observation of 5,000 lbs and 1805, the aircraft would consume at least 3,000 lbs of fuel. Further evidence of the flightcrew's lack of concern or awareness was provided when just after his observations of 4,000 lbs remaining about 17 min before the crash, the flight engineer left the cockpit at the captain's request to check on the cabin emergency evacuation preparations. Upon his return, about 4 min later, he gave the captain an estimate of another 2 or 3 min for the completion of the cabin preparation. At this time, the aircraft was in the general vicinity of the airport. In the initial interview with the captain, he stated that he felt the cabin preparation could be completed in from 10 to 15 min and that the "tail end of it" could be accomplished on the final approach to the airport. Certainly there was nothing more to do in the cockpit. All of the landing gear check procedures, as prescribed in the approved flight manual and recommended by company line maintenance, had been completed and dispatch had been notified and had alerted Portland company personnel of the problems.

Under these circumstances, there appears to have been no valid reason not to discontinue their heading inbound toward the airport in order to make their previously estimated landing time. However, about 1801:12, the first officer accepted and the captain did not question a vector heading which would take them away from the airport and delay their landing time appreciably. Moreover, after the turn was completed none of the flightcrew suggested turning toward the airport. Thus, it was at this time that the crew's continuing preoccupation with the landing gear problem and landing preparations became crucial and an accident became inevitable.

The Safety Board also considered the possibility that the captain was aware of the fuel quantity on board, but failed to relate the fuel state to time and distance from the airport and intentionally extended the flight to reduce the fuel load in order to reduce the potential of fire should the landing gear fail upon landing. The Safety Board could find no evidence, however, to support such a theory and believes that had he so intended, the captain would have advised the first officer and the flight engineer. Therefore, the Safety Board can only conclude that the flightcrew failed to relate the fuel remaining and the rate of fuel flow to the time and distance from the airport, because their attention was directed almost entirely toward diagnosing the landing gear problem. Although on two occasions the captain confirmed with the company that he intended to land about 1805 and that he would be landing with about 4,000 lbs of fuel, this estimated time of arrival and landing fuel load were not adhered to, nor was the expected approach time given to Portland Approach. This failure to adhere to the estimated time of arrival and landing fuel loads strengthens the Board's belief that the landing gear problem had a seemingly disorganizing effect on the flightcrew's performance. Evidence indicates that their scan of the instruments probably narrowed as their thinking fixed on the gear. After the No. 4 engine had flamed out and with the fuel totalizer indicating 1,000 lbs, the captain was still involved in resetting circuit breakers to recheck landing gear light indications.

It was not until after it became apparent to the crew that total engine flame out was imminent that the captain was concerned and, in fact, may have been confused as to the amount of fuel which actually remained. About 6 min before all engines stopped, the captain stated that there was 1,000 lbs of fuel in the No. 1 main tank, and the flight engineer agreed with him. At this same time, the captain began to describe the gage indication as changing from 1,000 lbs to zero lbs. Since the No. 1 main tank gage does not change its indication from 1,000 lbs to zero lbs directly, but decreases in increments of 100 lbs, the captain must have read the gage indication incorrectly. Actually, the action he described was that of a gage changing from 100 lbs to zero lbs.

The company had recently changed the fuel quantity gages on this aircraft from a direct reading digital-type to a three-figure indicator that had to be multiplied by a factor of 100 to get the actual individual tank values. In addition, the new totalizer gage, of the same three-figure presentation as the individual tank gages, had to be multiplied by a factor of 1,000 to get the actual total fuel. During the stressed situation, the captain and the flight engineer may have mixed up these multipliers and used 1,000 when reading the individual tank gages instead of 100. However, there is no evidence from previous comments that such a mistake was made. By the time such confusion was indicated, the accident was inevitable.

The Safety Board believes that this accident exemplifies a recurring problem--a breakdown in cockpit management and teamwork during a situation involving malfunctions of aircraft systems in flight. To combat this problem, responsibilities must be divided among members of the flightcrew while a malfunction is being resolved. In this case, apparently no one was specifically delegated the responsibility of monitoring fuel state.

Although the captain is in command and responsible for the performance of his crew, the actions or inactions of the other two flight crewmembers must be analyzed.

Admittedly, the stature of a captain and his management style may exert subtle pressure on his crew to conform to his way of thinking. It may hinder interaction and adequate monitoring and force another crewmember to yield his right to express an opinion.

The first officer's main responsibility is to monitor the captain. In particular, he provides feedback for the captain. If the captain infers from the first officer's actions or inactions that his judgment is correct, the captain could receive reinforcement for an error or poor judgment. Although the first officer did, in fact, make several subtle comments questioning or discussing the aircraft's fuel state, it was not until after the No. 4 engine flamed out that he expressed a direct view, "Get this . . . on the ground." Before that time, the comments were not given in a positive or direct tone. If the first officer recognized the criticality of the situation, he failed to convey these thoughts to the captain in a timely manner.

The flight engineer's responsibility, aside from management of the aircraft systems, is to monitor the captain's and first officer's actions as they pertain to the performance of the aircraft, that is, takeoff, landing, holding speeds, and range of the aircraft considering time and fuel flow. Although he informed the captain at 1750:30 that an additional "fifteen minutes is really gonna run us low on fuel here," there is no indication that he took affirmative action to insure that the captain was fully aware of the time to fuel exhaustion. Neither is there an indication that, upon returning to the cockpit at 1801:39, he relayed any concern about the aircraft's fuel state to the captain. Although he commented that 3,000 lbs of fuel remained, he failed to indicate time remaining or his views regarding the need to expedite the landing.

The first officer's and the flight engineer's inputs on the flight deck are important because they provide redundancy. The Safety Board believes that, in training of all airline cockpit and cabin crewmembers, assertiveness training should be a part of the standard curricula, including the need for individual initiative and effective expression of concern.

In order to determine whether the captain had received all available assistance during the emergency, the Safety Board evaluated the actions of the company dispatcher and his role relative to the accident sequence. According to the tape of the conversation between the captain, the company dispatcher, and company line maintenance personnel, the captain had advised the dispatcher that he had 7,000 lbs of fuel aboard and that he intended to land in 15 or 20 min. The dispatcher then checked with the captain to ascertain a specific time for the landing and the captain agreed that 1805 was "a good ballpark." The dispatcher, according to his interview after the accident, then relayed this landing time and the aircraft's status to the company personnel in Portland. He also stated that his assessment of the situation was that of the fuel remaining upon landing would be low but the landing could be made successfully at 1805. The Safety Board believes that, with the information given to him by the captain, the dispatcher acted properly and in accordance with company procedures.

3. CONCLUSIONS

3.1 Findings

1. The flightcrew was properly certificated and qualified for the flight.

2. The aircraft was certificated, maintained, and dispatched in accordance with Federal Aviation Regulations and approved company procedures.

3. Except for the failure of the piston rod on the right main landing gear retract cylinder assembly, with the resulting damage to the landing gear position indicating system switch, there was no evidence of a failure or malfunction of the aircraft's structure, powerplants, flight controls, or systems.

4. The aircraft departed Denver with the required fuel aboard of 2 hrs 26 min for the en route flight and with the required FAR and company contingency fuel aboard of about 1 hr.

5. The aircraft began holding about 1712 at 5,000 ft with its gear down; this was about 2 hrs 24 min after it departed Denver.

6. The landing delay covered a period of about 1 hr 2 min.

7. All of the aircraft's engines flamed out because of fuel exhaustion about 1815—1 hr 3 min after it entered into hold and 3 hrs 27 min after it departed Denver.

8. Fuel exhaustion was predictable. The crew failed to equate the fuel remaining with time and distance from the airport.

9. No pertinent malfunctions were found during examinations of the fuel-quantity measuring system.

10. A new digital fuel-quantity indicating system was installed on this aircraft on May 12, 1978. This was in accordance with a DC-8 UAL fleetwide retrofit program.

11. Evidence indicates that the fuel quantity indicating system accurately indicated fuel quantity to the crew.

12. The fuel gages are readily visible to the captain and the second officer.

13. The captain failed to make decisive timely decisions.

14. The captain failed to relate time, distance from the airport, and the aircraft's fuel state as his attention was directed completely toward the

diagnosis of the gear problem and preparation of the passengers for an emergency landing. The gear problem had a disorganizing effect on the captain's performance.

15. Neither the first officer nor the flight engineer conveyed any concern about fuel exhaustion to the captain until the accident was inevitable.

3.2 Probable Cause

The National Transportation Safety Board determined that the probable cause of the accident was the failure of the captain to monitor properly the aircraft's fuel state and to properly respond to the low fuel state and the crewmember's advisories regarding fuel state. This resulted in fuel exhaustion to all engines. His inattention resulted from preoccupation with a landing gear malfunction and preparations for a possible landing emergency.

Contributing to the accident was the failure of the other two flight crewmembers either to fully comprehend the criticality of the fuel state or to successfully communicate their concern to the captain.

4. Safety Recommendations

As a result of this accident, the Safety Board has issued the following recommendations to the Federal Aviation Administration:

> "Issue an Operations Alert Bulletin to have FAA inspectors assure that crew training stresses differences in fuel-quantity measuring instruments and that crews flying with the new system are made aware of the possibility of misinterpretation of gage readings. (Class II--Priority Action) (A-79-32)"
>
> "Emphasize to engineering personnel who approve aircraft engineering changes or issuance of Supplemental Type Certificates the need to consider cockpit configuration and instrumentation factors which can contribute to pilot confusion, such as the use of similar-appearing instruments with different scale factors. (Class II--Priority Action) (A-79-33)"
>
> "Audit Supplemental Type Certificate SA3357WE-D for completeness, especially in the area of system calibration after installation. (Class II--Priority Action) (A-79-34)"
>
> "Issue an operations bulletin to all air carrier operations inspectors directing them to urge their assigned operators to ensure that their flightcrews are indoctrinated in principles of flightdeck resource management, with particular emphasis on the merits of participative management for captains and assertiveness training for other cockpit crewmembers. (Class II, Priority Action) (A-79-47)"

ICAO Note: Appendices A, B, C and D to the report are not reproduced.

ICAO Ref: 442/78

No. 3

Boeing 727-200, JY-ADU, accident at Doha International Airport, State of Qatar, on 13 March 1979.
Report No. 1/79, dated 15 December 1979, *released by the State of Qatar.*

SYNOPSIS

The accident was notified by Doha International Airport air traffic services on 13 March 1979.

The Civil Aviation Department carried out an investigation and the following groups were established:

Operations; Engineering; Flight Recorder; Witness;
Meteorological; Air Traffic Services; Airport Fire Service.

The States of Registration and Manufacture were notified of the accident and accredited representatives were appointed to participate in the investigation.

The accident happened at night when the aircraft was executing a missed approach procedure on Runway 34 at Doha International Airport. Following the initiation of the missed approach procedure and on reaching a height of approximately 750 ft above aerodrome level the aircraft encountered a thunderstorm the effects of which exceeded the climb performance of the aircraft and forced it to the ground.

Note.- *Technical Services at Doha International Airport, including Air Traffic* Services, Meteorological Services, Fire and Rescue Services, Motor Transport Services and Airport Electrical and General Maintenance Services, are provided by International Aeradio Limited under contract to the Government of the State of Qatar.

1. FACTUAL INFORMATION

1.1 History of the Flight

The Boeing 727 aircraft registration JY-ADU, using call sign Royal Jordanian 600, was operating a scheduled passenger service from Amman to Doha and Muscat.

With the First Officer as the handling pilot the aircraft left Amman at the scheduled time of 2155 on 13 March 1979 with a total of 64 on board. The first part of the flight was uneventful. As it drew near Doha the aircraft encountered turbulence.

R/T contact was established with Doha at 2308 at a range of 45 miles when the controller passed on the following weather information:

Wind velocity 090°/17 kt, visibility 10 km in thunderstorms, 1/8 CB at 2 500 ft, 3/8 at 3 000 ft, QNH 1008. The controller added that the thunderstorm appeared to be building up over and to the north west of the airfield.

After some R/T exchanges as to which runway was preferable, the pilot requested Runway 16 and was cleared for a visual approach to that runway or for a NDB/VOR approach if visual contact with the airfield was not established. The aircraft reported approaching the overhead position for a NDB/VOR procedure at 2317 and was seen by the controller as it positioned for the outbound leg of the procedure. On the airfield rain was falling quite heavily and the controller informed the aircraft of this. At this stage the NDB tripped and the controller informed the aircraft accordingly adding that, in his opinion, this was in consequence of a lightning strike. At 2322.30 the controller advised that the NDB was back on the air. Lightning was intense and continuous close to the airfield to the north and north-west.

At 2325 the aircraft reported procedure turn complete and was cleared to land. In the course of the approach the following winds and information were passed to the aircraft:

Wind 010°/08 kt-backing; 340°/06 kt; 340°/less than 5 kt with heavy rain; 290°/14 kt.

At 2329 the aircraft reported that it had not seen the runway, and initiated a missed approach, requesting and receiving clearance to make an ILS approach to Runway 34. At 2330 the controller gave the aircraft a wind of 280°/16 kt and at 2331 a wind of 290°/14 kt, a QNH of 1011 mb, and a report of heavy rain. At 2335 the aircraft reported procedure turn complete and field in sight. Landing clearance was given with a wind of 180°/06 kt.

During the next 2 minutes of the final approach the controller reported winds of 150°/13 kt, 150°/15 kt, and 140°/17 kt; he also gave the pilot an "unofficial" report (i.e. not based on an observation by a meteorological observer) that the rain was very heavy and that visibility from the tower was probably less than 1 000 m.

Note.- All times in this report are GMT. Local time is GMT plus 3 hours.

Because of the increasing tail wind component and difficulties in maintaining the glide path both pilots simultaneously decided, before reaching decision height, that a missed approach was necessary and this was initiated at 2337.19 at a height of approximately 300 ft above aerodrome level (a.a.l.). The pilot then requested clearance to Dahran and the controller cleared the aircraft to climb straight ahead to FL 80.

In accordance with Flight Manual procedures "go around" power was applied and established: 25° flap was selected: the undercarriage was raised and the "go around" attitude of 12° established, producing an average rate of climb approaching 1 300 ft/min. The flight data recorder shows that the aircraft climbed to 750 ft a.a.l. with the indicated airspeed (IAS) falling to approximately 140 kt. The aircraft then began to descend at an increasing rate until it struck the ground with a vertical speed of 4 200 ft/min., with the IAS increasing rapidly to approximately 170 kt in the last 5 s. The cockpit voice recorder shows that in the last 11 s before impact the Ground Proximity Warning System (GPWS) gave a continuous aural warning, and that during this period the crew increased power from the "go around" setting to maximum. The aircraft struck the ground on the left-hand edge of the runway, 2 050 m from the threshold with the nose approximately 10° above the horizon and with the left wing 5° down.

The aircraft "bounced" off the first impact and turning left, rolled to the right to an inverted position and then slid, tail first, into the Fire Station garage some 800 m from the first impact point and 270 m from the runway edge. The fuselage broke into three main sections during the process.

The nose and centre sections were destroyed by fire but the tail section remained untouched by fire.

1.2 Injuries to Persons

Injuries	Crew	Passengers	Other
Fatal	4	41	
Non-Fatal	9	6	
None	2	2	
Total:	15	49	

The First Officer and the Flight Engineer survived the accident.

1.3 Damage to Aircraft

The aircraft was destroyed by impact and fire.

1.4 Other Damage

Three runway and one taxiway light fittings were broken and a marker board alongside the runway designating the displaced daylight threshold of RW 16 was demolished.

A six-bay garage and store building together with miscellaneous stores and equipment were destroyed by impact and fire.

Windows in the Fire Station and in the Fire Services Watchroom were broken by blast and flying debris.

The front facet of the Fire Station suffered minor impact damage.

One water tanker, one emergency tender, one ambulance and one passenger bus were destroyed. One foam tender suffered considerable fire damage.

1.5 Personnel Information

1.5.1 Flight Crew

The Captain, aged 32, held Airline Transport Pilot Licence No. 371, issued by the Kingdom of Jordan on 18 May 1977. It was valid until 6 July 1979 and was endorsed for pilot-in-command on Boeing 727 aircraft. His last instrument rating and proficiency check was carried out on 5 January 1979. His last medical examination was 15 October 1978, when he was assessed as fit. He had flown a total of 4 606 hours, of which 102 were in command on B 727's. In the thirty days prior to the accident he had flown a total of 50 hours 05 minutes. Immediately prior to the accident he was on standby from 1000 to 2200 on 11 March and had a rest day on 12 March. He was authorized to give First Officer landings and take-offs, in accordance with Company Orders, on 10 March 1979.

The First Officer, aged 38, holds Commercial Pilot Licence No. 418 issued by the Kingdom of Jordan on 26 May 1975. It is valid until 27 March 1979 and is endorsed for Co-pilot on B 727 aircraft. His last instrument rating and proficiency check was on 27 September 1978, and his last medical examination was on 30 January 1979, at which examination he was assessed as fit. He has flown a total of 2 674 hours. In the 30 days prior to the accident he had flown a total of 72 hours 05 minutes. Immediately prior to the accident he had flown 4 hours and 05 minutes on 11 March and had a rest day on 12 March.

The Flight Engineer, aged 24, holds Flight Engineer Licence No. 107 issued by the Kingdom of Jordan on 4 February 1978. It is valid until 16 April 1979 and is endorsed for Flight Engineer on Boeing 727 aircraft. His last proficiency check was on 16 August 1978 and his last medical examination was on 7 March 1978, at which examination he was assessed as fit. He has flown a total of 734 hours. In the 30 days prior to the accident he had flown 39 hours 55 minutes. Immediately prior to the accident he had been on Rest Day on 10 March and standby from 2200 on 11 March to 1000 on 12 March.

1.5.2 Cabin Crew Staff

The cabin crew staff had last taken courses on emergency drills and procedures as follows:

Purser	20/5/78 - 10/6/78
Cabin Attendant	17/7/78 - 30/7/78
Cabin Attendant	13/1/79 - 18/1/79
Cabin Attendant	20/1/79 - 25/1/79
Cabin Attendant	20/1/79 - 25/1/79
Cabin Attendant	16/11/78 - 21/11/78

There were 6 extra crew members on board who were seated throughout the cabin and treated as normal passengers. They had no special safety or survival training, or function within the operational crew framework.

1.6 Aircraft Information

1.6.1 History of Maintenance Synopsis

The aircraft, a Boeing 727-200, was manufactured by the Boeing Aeroplane Co. during 1974 and given the serial number 20886. It was registered in the Kingdom of Jordan by the owner, Royal Jordanian Airline Corporation, on 31 July 1974 and issued with the nationality and registration mark JY-ADU; its number in the Jordanian Aircraft Register was 194. A Certificate of Airworthiness first issued on 10 August 1974 was renewed on 27 September 1978 valid for twelve months.

The aircraft was maintained to an approved maintenance schedule requiring progressive inspection at 200-hour intervals. On 12 March 1979 a maintenance check including schedule items A check, B2 check, C check and Phase F was completed prior to service, at which time the aircraft had accumulated 10 237.30 hours of flight time.

There were no significant defects recorded as outstanding prior to the aircraft's being released for service on the flight during which the accident occurred.

Fuel used in the aircraft was Jet A1.

1.6.2 Weight and Balance

The load sheet for the flight was examined together with the General Declaration Form, passenger and cargo manifests, trim sheet and Captain's load information, and loading instruction form. The load sheet showed a dry operating weight of 47 461 kg, weight of fuel at take-off 19 700 kg, 43 adult passengers and one infant totalling 3 410 kg, baggage 960 kg, cargo 4 654 kg and mail 26 kg. Take-off weight was therefore 76 211 kg; this, minus trip fuel of 9 750 kg, gave a predicted landing weight of 66 461 kg. The maximum authorized landing weight with 30 flap was 72 575 kg and with 40 flap was 64 637 kg. The take-off and zero fuel weight trim settings were within limits.

The Captain's load information did not show a 146 kg consignment of flammable liquid stowed in hold no. 2.

1.7 Meteorological Information

1.7.1 Synoptic Weather Situation: Arabian Gulf

Relatively high surface pressure over Iran, and low level easterly air stream over Saudi Arabia maintained a moist low level easterly air stream over Qatar, and low level convergence over the central and western Gulf produced large CB clouds and thunderstorm activity.

1.7.2 Weather at Doha International Airport

Following relatively clear weather during the morning of 13 March, thunderstorm activity commenced at 1550 at the airport and continued intermittently throughout the afternoon and evening.

The 2257 weather report to ATC indicated thunderstorm activity. Special weather reports at 2329 and 2336 reported thunderstorms which appeared to the meteorological observers to pass directly over the airport. Prior to and at the time the accident occurred, the rainfall was recorded as "Violent". Subsequent analysis of the rain recorder trace at the airport shows that 29 mm of rain fell between 2320 and 2350, which exceeds an equivalent rainfall rate of 50 mm/h. The Captain and First Officer of an aircraft parked on the apron waiting for the weather to clear before taking off both testified that it was the worst storm they had ever seen on the ground - this was at the time JY-ADU had started outbound for an ILS approach on RW 34.

The First Officer aboard JY-ADU, in his evidence, said that at the start of the missed approach off RW 34 there was turbulence and heavy rain.

The airport barogram record shows that the pressure was 1010 mb at 2138 and fell steadily to 1008 mb at 2323. Pressure rose suddenly to 1012 mb between 2323 and 2329 then remained between 1011 and 1012 mb until 2400 and fell slowly thereafter.

From 2317 until 2342 marked changes in wind speed and direction occurred as recorded in paragraph 1.1.

The accident occurred at night.

1.7.3 Weather Information Available to the Flight Crew

The following information regarding Doha weather was given to the aircraft prior to take-off at Amman:

TAF 1800 - 0300, WIND 080/15, VIS 10 KM PLUS
CLOUD 2/8 CU 3000, 3/8 AC 9000
PROBABILITY 30% THUNDERSTORM
TEMPO WIND GUSTING TO 35 KTS
VIS 3000 MTR IN THUNDERSTORM AND
MODERATE OR HEAVY RAIN
METAR 1500Z, WIND 080/14, VIS 10 KM PLUS,
CLOUD 2/8 CU 2500, 1/8 CB 3000, 3/8 AC 9000.
TEMPERATURE 21, DEWPOINT 18
QNH 1010.9

SIGMETS: weather of thunderstorm activity in the area of the FIR Bahrain extending to 23° south, were in operation throughout the period of the flight of the aircraft.

1.8 Aids to Navigation

At 2317 the NDB tripped off and was reset at 2322.

At 2330 the VOR went off the air and was not brought back into service until after the accident, when it was ascertained that the equipment was serviceable but had apparently tripped off due to transients on the power line caused by lightning strikes.

It is believed that the NDB and VOR outages are not relevant to the accident and this is confirmed by the evidence of the First Officer.

The instrument landing system on RW 34 operated normally and the routine flight check on 6 March, as well as a post-accident flight check on 17 March, indicated that the equipment operated within ICAO Cat II tolerances.

1.9 Communications

Two way communications on 118.9 MHz were established and maintained with the aircraft throughout its approach to Doha.

1.10 Aerodrome Information

RW 34/16 has an asphalt surface and is 4 572 m long and 46 m wide. It has green, flush threshold lights and high-intensity bi-directional and low-intensity omnidirectional edge lights. Approach lighting on RW 34 is a variable high-intensity Calvert system, and on RW 16 a variable low-intensity centre line with crossbar system. T-VASIS are installed on both ends of the runway. The runway and its systems were serviceable at the time of the accident.

The meteorological observations are taken from the roof adjacent to the control tower except for the surface wind which is sensed by a 30 ft high anemometer 600 m SSE of the control tower. The control tower and fire station are approximately 3 000 m from the threshold of RW 34 and hence approximately 4 500 m from an aircraft on the glide slope at an altitude of 300 ft.

1.11 Flight Data Recorders

The flight data recorder (FDR), Fairchild type 5424 - 221, serial 2149 and the cockpit voice recorder (CVR), Collins 642 C-1, serial No. 2283 located in the rear portion of the aircraft were recovered undamaged.

The records were transcribed by the UK Accident Investigation Branch and a graphical representation of the FDR data recorded in the final 60 s before impact is annexed to this report.

The US National Transportation Safety Board provided an analysis of the FDR data comparing the rate of climb and acceleration with predicted performance levels of the aeroplane.

Datum corrections were applied in the use of the readouts in the investigation.

1.12 Wreckage

1.12.1 Impact Details; Airframe and Systems Damage

The aircraft first struck the ground on the left edge of RW 34 approximately 2 050 m from the threshold, on a track of 330°M. The first impact occurred with the underside of the rear fuselage at a position on and forward of the aft passenger stairway. The left wing tip and the left wing leading and trailing edge flaps struck at the same time. There was severe damage to the fuselage, aft cabin windows were broken and the rear baggage hold burst open. Both podded engines (nos. 1 and 3) became detached. The first impact damage and ground markings indicated an impact attitude of about 10° nose up and 5° left wing low. The impact configuration was established as landing gear up and locked, inflight and ground spoilers retracted, leading edge slats and flaps extended, trailing edge flaps at or close to the 25° position.

Following the first impact the aircraft "bounced" and reimpacted the ground about 100 m further on, the left wing demolishing the inside marker board for the displaced threshold of RW 16 to the west of the runway. The aircraft then swung to the left and skidded sideways down Taxiway 4. In the area of the intersection of Taxiways 4 and 5 the aircraft rolled to the right to become inverted, causing substantial damage to the right wing and detaching the right wing slats. An impact on the outboard upper surface of the right horizontal stabilizer fractured the horizontal and vertical stabilizers and removed the upper rudder. The major part of the wreckage then slid into the Fire Station garage, the fuselage lying on its right side and travelling rearwards. The fuselage came to rest in three sections; the flight deck and about 14 ft of forward fuselage, a 53 ft section of centre fuselage with wing roots and the remaining tail section.

Fire started at the time the aircraft struck the Fire Station garage. The fire eventually consumed much of the centre fuselage section and inboard mainplanes. The tail section remained untouched by the fire; the flight deck section interior was destroyed.

There was no evidence from examination of the wreckage or its distribution, of airborne structural failure or detachment.

The total disruption due to impact and fire precluded much direct assessment of aircraft system integrity prior to impact; however, evidence from the First Officer and the Flight Engineer coupled with FDR and VCR readouts indicate that systems functioned normally.

There was no evidence of airborne fire or lightning strike.

1.12.2 Engines

Neither the First Officer nor the Flight Engineer noted any engine malfunction prior to impact, and the CVR readout contains no comments by the flight crew about possible power plant failure.

Under the control of the U.S. National Transportation Safety Board the engines were stripped and examined, particularly to determine if performance might have been impaired by water ingestion. The NTSB report concludes that:

> "All of the observed damage in the three engines is indicative of high power operation at ground impact and breakup of the aircraft. Especially noteworthy is the amount of sand deposited throughout the engines. If the engines were at low power, or not operating, when they struck the ground, the sand would not have been found throughout the engine.
>
> "Particularly it would not have been found in such great quantity in the diffuser and combustion sections. The sand could only have collected and packed in those areas and the bleed system during high air flow, and could only have caked on the turbine vanes and blades when they were at operating temperature.
>
> "In addition, the molten metal splatter on the turbine nozzle vanes and turbine blades could only have melted and adhered when those parts were at operating temperature. There is no doubt that this metal came from airframe debris ingested by the engines and from pieces of compressor blade broken out by that debris.

"All the compressor damage noted is foreign object damage, and there is no damage which can be associated with interference between compressor rotor blades and stator vanes. Past experience with the JT8D-17 engine has shown that at extremely high rates of water ingestion under part power conditions, a severe stall may occur which could cause blade-to-vane interference. There is no evidence that such interference occurred in these engines. All the physical indications are that blade clashings did not occur.

"There was nothing found in the engines to indicate that water ingestion affected their operation. All evidence indicates the engines were operating at high power at impact."

1.13 Medical and Pathological Information

There is no suggestion or evidence to suggest that flight crew incapacitation was a factor in the accident.

Of the forty-five persons who died in the accident it was ascertained that the main cause of death in the case of twenty of them was severe burns and the remaining twenty-five suffered fatal impact injuries.

1.14 Fire

1.14.1 Equipment and Organization

At the time of the accident the main strength of the Airfield Fire Service, consisting of five vehicles under the command of the Duty Fire Officer, was at the "bad weather standby" position for RW 34. The position is approximately abeam the touchdown point of RW 34 and is intended to give an improved response time in the event of a landing accident, as well as assisting ATC by allowing the Fire Service to give information about landing aircraft. The remaining fire vehicle in use was in position at the Fire Station which is close to ATC, about 3 000 m from the threshold of RW 34. All equipment in use was serviceable; scales of equipment and manning levels met the requirements of ICAO Aerodrome Category 7, which includes use of the airport by B-727 aircraft.

In general the standard of English spoken in the Fire Service was not high. Routine messages needed to be expressed carefully, and an acknowledgement was not always proof that a message would be correctly understood. On the night of the accident it was noteworthy that both the Watchroom and the Duty Fire Officer followed correct R/T procedures.

The Fire Service Watchroom and all vehicles were equipped with two-way radio equipment which enabled them to communicate on 121.3 MHz, the ground movement frequency (GMF). This was partially ineffective at the time of the accident, although the details were not obvious to users of the frequency. An analysis of transmissions shows that two-way communication existed between ATC and the Fire Service Watchroom, and also between ATC and vehicles at the "bad weather standby" position for RW 34.

However, communications between the Watchroom and the deployed vehicles was intermittent. It was not possible to establish a positive reason for this. It is considered, however, that the impact of the wreckage against the Watchroom, the failure of a radio component in the Watchroom through high levels of lightning activity, the presence of large areas of standing water on the airfield, the screening effect of high-sided vehicles, or a combination of these circumstances may have caused the poor communications.

1.14.2 Fire Fighting Action

The main portion of the wreckage was the fuselage, which slid backwards and sideways, practically upside down, into the Fire Station garage/workshop, breaking into three parts. The larger part, the fuselage centre section, struck vehicle parking bays and the workshop contained cylinders of oxygen and acetylene, together with drums of lubricating oil and other flammable materials.

The impact knocked down several pillars which supported the roof of the vehicle parking bays, bringing rubble and building materials down on the centre section, and also pushed three heavy motor vehicles sideways into each other, demolishing the intervening walls. There was an explosion and an eruption of flame either as the aircraft struck the Fire Station, or very shortly afterwards. The tape recording shows that the Watchroom attendant called "Fire, fire, fire, Watchroom, fire", some seventeen seconds after the aircraft struck the runway. There is some evidence that fire may have begun near the rear end of the broken centre section.

Fuel quickly began to spread around the centre section and to run downhill, away from the Fire Station, clear of the nose and tail sections.

Rescue 6, the vehicle at the Fire Station, closed promptly with the fire in the first few moments after the accident, but the Leading Fireman in charge saw that there were people still alive and in need of assistance. Accordingly he and one of his men went to help, leaving two men to operate the vehicle. The Leading Fireman decided to cease operating the main foam delivery monitor and to use one side hose, so that its lower rate of delivery could sustain protection for the rescue operation over a longer period. Professional opinion is that the Leading Fireman made a sound decision in difficult circumstances. The Duty Fire Officer, with his five vehicles, did not receive positive instructions that there was an accident and that he was needed at the Fire Station until approximately eight minutes after the impact, when the crew of Rescue 6, realizing that they were not being reinforced, called the Duty Fire Officer and his five vehicles to the accident site. It is estimated that a total time of eleven minutes must have elapsed before the Duty Fire Officer and the main force of vehicles were in attendance at the scene of the accident.

The rain was so heavy that, from the control tower, the controller could not visually confirm that the aircraft had crashed into the Fire Station, about 100 m away. The Watchroom had been struck by the wreckage, but the broadcast of "Fire, fire, fire, Watchroom, fire" by the Watchroom attendant was not heard by the Duty Fire Officer. The Duty Fire Officer made attempts to find out what was happening, but to him the situation was confused because he could hear the controller transmitting to the Watchroom, but could not hear the answers. One transmission from the Duty Fire Officer received a hurried reply from the controller who did not wait for an acknowledgement before continuing a conversation with the Watchroom; another request to ATC for information was blocked by other fire vehicles as they started discussing, in their own language, the problems of their vehicles in the heavy rain. A third request, addressed to the Watchroom, produced no answer, either because of the communications problem, or because it was made after the Watchroom was abandoned owing to fumes and heat.

The controller assumed that the Duty Fire Officer had heard the Watchroom cry of "Fire, fire, fire, Watchroom, fire," and, believing that the Watchroom knew more about the situation than he did, took it for granted that the Watchroom would summon the Duty Fire Officer. The orders for controller and Watchroom do not make it clear whose responsibility this was. In fact, the Watchroom attendant took no such action.

When members of the Fire Service conversed in their own language the controller took this as a sign that they were in action at the fire. The person who eventually summoned the Duty Fire Officer at 2436 was a junior fireman, the driver of Rescue 6.

On arrival at the scene of the fire at an estimated time of 2349, the vehicles from the "bad weather standby" position commenced fire control and rescue operations immediately. The extraordinarily heavy rain diluted the foam, and the building materials which were lying on the fuselage centre tended to deflect foam, preventing it from lying on the aircraft itself.

At about 2359, when rescue work was well in hand and the Fire Service thought the fire was under control, there was a loud explosion near the front end of the fuselage centre section. This explosion was sufficiently violent to send metal flying through the air and blew in the remaining windows of the Watchroom. It was accompanied by a renewed outbreak of fire, and was followed by a further explosion at approximately 0014 hrs, less violent in nature. The fire gradually spread to the nose section and was finally brought under control at approximately 0305 hrs. Its is considered that the Fire Service performed commendably once all vehicles had arrived at the site of the fire.

The Doha town fire and ambulance services responded immediately to the '999' call made by ATC as part of the post-accident procedures. The appliances arrived at the accident site some 15 minutes after the call.

1.15 Survival Aspects

Forty-one dead bodies were recovered from the wreckage. Fourteen persons were rescued alive by the Fire Service, four of whom subsequently succumbed to injuries. Nine persons got out of the aircraft by their own efforts.

Of the forty-five deaths, twenty were found to be as a consequence of the effects of fire and twenty-five were due to impact injuries.

The Captain of the aircraft died as a result of impact injuries and was dead when the aircraft came to a stop.

The survivors were recovered in about equal proportions from the front part of the main fuselage and from the tail section. The First Officer was assisted from the flight deck by the Duty Fire Officer in Rescue Vehicle 1. At this time the flight deck was not on fire. Two cabin attendants escaped through a break in the forward fuselage and at least one passenger pushed a window out and escaped through that. Information regarding the evacuation of other persons is of a fragmentary and unco-ordinated nature and of no particular value in assessing the situation.

The aircraft struck the ground initially with a rate of descent of over 4 000 ft/min at a forward speed of over 160 kt. Before coming to rest, the aircraft rolled and turned, striking the ground at least two more times. Some seats became detached from their fastenings and others were severely distorted.

Fire became well established at impact with the Fire Station garage and was immediately attacked by major foam appliance Rescue 6 by monitor-dispensed foam. However, the foam application was, quite correctly, reduced to the use of one side hose to permit the firemen to carry out passenger rescue operations. It has not been possible to assess the effect of this limitation on the progress of the fire.

Rescue Vehicles 1, 2, 4, 5 and 8 arrived at the accident scene approximately 11 minutes after the start of fire.

1.16 Tests and Research

In order to establish the accuracy of witness statements, a varying series of missed approaches were flown by a heavy jet aircraft. During this sequence, the witnesses consistently identified a particular type of approach and missed approach as similar to the one they had seen performed by RJ 600. This correlation supported statements by surviving crew members and the information later derived from the FDR.

1.17 Additional Information

The ALIA Flight Operations Manual contains comprehensive guidance material concerning wind shear and thunderstorm activity and the following extracts are quoted as relevant to this accident:

Wind Shear

"Wind shear is best described as an abrupt change in wind direction and/or velocity in a very short distance in the atmosphere. Under certain conditions, the atmosphere is capable of producing some dramatic shears very close to the ground. For example, wind direction changes of 180° and velocity changes of 50 kt or more within 200 ft of the ground have been observed. However, this is not something that is encountered every day; in fact, the unusualness of the wind shear phenomenon makes it all the more a problem.

"The most common cause of significant low-level wind shear is the gust front associated with a thunderstorm. Although this front may be as much as 10 miles ahead of the thunderstorm, a clue to the flight crew that wind shear may be present is the thunderstorm itself.

"The worst possible occurrence in a shear is to fly into a sudden tailwind with its attendant loss of lift and airspeed. Downwind landings, certain noise abatement climbout headings, and similar manoeuvres can put an airplane in this situation when shear is present.

"The probability of encountering wind shear during the approach should be assessed as described previously. The Captain should, however, supplement this information with a comparison of the latest wind directions and velocities at various stations in the vicinity of the destination.

"In the terminal area and during the approach, moreover, the Captain should monitor the INS winds and compare them with the surface winds for a low-level wind shear indication. When the wind from the INS is not available during the approach, the difference of INS ground speed and the TAS should be compared with the surface headwind or tailwind component as a measure of shear. The Captain should: (1) brief the flight crew on possible shear conditions,

(2) check if runway length permits use of reduced landing flap setting, (3) add an increment of speed to the normal approach speed, (4) back up the autothrottles manually or use manual thrust control, and (5) instruct the crew to monitor thrust, pitch attitude, vertical speed, IAS, and turbulence.

"During the approach the required thrust and vertical speed are the most important and easily recognized parameters associated with the wind shear phenomenon.

"The most hazardous consequences occur when a wind shear occurs close to the ground after thrust adjustments have been already made during the approach to compensate for wind".

Thunderstorm activity

"In the mature stage, the mass of water, snow and hail increases to the extent that it can no longer be supported by the drafts. The resulting precipitation causes downdrafts which can be very strong and turbulent.

"The dissipating stage occurs after the downdrafts and precipitation have spread and the updrafts have disappeared. Turbulence is much less intense than in the preceding stages. Every thunderstorm has one or more separate cells of updrafts and downdrafts with turbulence of varying intensity. The thunderstorm cloud represents only the visible portion of the updraft and downdraft areas. Turbulence from these drafts also exists outside the cloud area.

"The greatest turbulence is associated with the areas of heaviest rain concentrations. Severe turbulence is usually encountered between altitudes of 12 000 and 20 000 ft but may be found at any level. Moderate to severe turbulence can also be evident up to 10 000 ft above the cloud tops when high wind conditions exist.

"The strong winds and low-level wind shears that precede thunderstorms are known as 'gust fronts' or 'plow winds'. Gust fronts can be extremely hazardous during the flight phases associated with takeoff and landing. Typical gust front structure and characteristics are:

A. A surface wind shift often accompanies the gust front but may lead the front by up to 5 miles.

B. Ground-level sustained winds and gusts to 50 kt are common; occasionally, winds approach 100 kt.

C. The gust front moves faster than the thunderstorm from which it was created and may lead the thunderstorm radar echo by 5 to 10 miles.

D. Airplane or ground radars may show the gust front as a thin line echo caused by differences between the high air density behind the front and the relatively low air density ahead of it.

E. The gust front may travel faster at approximately 2 500 ft above ground level than on the surface; therefore, it should be expected even further ahead of the thunderstorm than surface disturbances may indicate.

F. Vertical wind shears of 10 kt/100 ft extending from the surface to several hundred feet above the ground may occur.

G. Horizontal wind shears of 50 kt per mile have been measured while crossing perpendicularly through the gust front, and the shear may be even greater in thunderstorm squalls. [See "Wind Shear", this chapter.]

H. The pressure jump reported in hourly weather sequence reports is helpful in determining the position of the gust front. The pressure jump usually precedes the gust front by approximately three miles.

I. A pressure altitude increase of 200 to 300 ft is common in advance of the gust front."

2. ANALYSIS

2.1 Aircraft Operations

No evidence of aircraft mechanical failure came to light during the investigation, and the First Officer and the Flight Engineer confirm that the aircraft handled and performed normally throughout the flight.

The Captain had recently completed a course of training in the process of qualifying for command of the Boeing 727 and could be expected to have had extensive recent practice in "go around" techniques. In addition, all the available evidence, including FDR & CVR readouts, indicates that the crew correctly followed Flight Manual "go around" procedures and maintained the appropriate aircraft attitude, speed and configuration from initiation of the last missed approach to impact with the ground.

In consequence of the above, it is considered that neither mechanical failure nor mishandling by the flight crew caused the aircraft to descend to the ground.

However, the meteorological data indicates that there was severe thunderstorm activity at and in the vicinity of the airfield when the accident occurred, and it is concluded that the aircraft was forced to the ground by a downburst associated with the thunderstorm activity, the effects of which exceeded the climb capability of the aircraft by a catastrophic margin. The decision by the flight crew to continue the approach to land in known thunderstorm conditions had a direct bearing on the accident and this point is therefore considered.

The forecast of a 30 per cent probability of thunderstorms in the Doha area was made available to the flight crew before take-off from Amman.

At first contact with the aircraft, Doha ATC passed on the current METAR and advised of thunderstorm activity at and near the airfield. During the approaches to RWs 16 and 34, ATC passed reports of heavy rain and frequent wind changes to the aircraft. These reports, plus evident lightning and turbulence and a 3 mb pressure change in 30 minutes, should have alerted the flight crew regarding the thunderstorm activity.

Thorough and comprehensive briefing material regarding thunderstorms and wind shear is contained in the ALIA Flight Operations Manual. There is no indication that the many warnings and cautions in the Manual were taken into account when the approach to RW 34 was commenced.

However, based on the available evidence it is considered that the flight crew did not wilfully enter into an area of severe weather conditions, but rather failed to understand the significance of the warning signs which should have alerted them to the severity of the weather system and the possible consequences of attempting the approach.

The First Officer flew the aircraft throughout the flight and the possibility that this factor may have contributed to the accident was considered. The Captain held Company authorization to permit the First Officer to fly the aircraft during take-off and landing. Limitations on such authorization include:

a) the runway must be clean and dry;

b) the crosswind component shall not exceed 15 kt;

c) for any landing or take-off the maximum tailwind allowable shall be 10 kt.

If the First Officer had landed the aircraft off the last approach, limitations a) and c) would have been exceeded. However, a landing was not made and no technical infringment occurred. In addition, the CVR transcript shows that the Captain maintained a close monitoring function throughout the approach and was contemplating a missed approach during the descent on the ILS. Although the runway conditions were unsuitable for landings by a First Officer, there is no evidence to suggest aircraft mishandling by the flight crew and it is concluded that the fact that the First Officer was at the controls of the aircraft was not a contributory factor to the accident.

2.2 Fire Service Operations

Because the Duty Fire Officer did not receive positive instructions to the contrary he remained, together with his fire vehicle group, at the "bad weather standby" position for RW 34 until he was called to the accident site by the crew of Rescue 6. This call was made some eight minutes after the fire started and resulted in the main strength of the Fire Service arriving at the scene of the accident some eleven minutes after impact.

The division of responsibility between ATC and Watchroom regarding the co-ordination of Fire Service action when vehicles were deployed at "bad weather standby" positions was not defined in the orders for the Watchroom attendant and for the controller.

From a statement made by the controller, it is apparent that he believed that, not long after he had operated the crash alarm, a fully informed Watchroom had assumed responsibility for controlling the Fire Service and that he was free to follow post-crash procedures as laid down in his instructions. On the other hand, the Watchroom attendant, who had not been trained to regard it as his task to redeploy Fire Service vehicles once they were upon the airfield, took no further action after calling "Fire, fire, fire, Watchroom, fire".

The basic reason for the delay in summoning the main strength of the airfield Fire Service therefore occurred because of the controller's assumption that the Watchroom attendant was exercising control over fire vehicles when this was not in fact the case. The delay was extended because of the communications problem between the Watchroom and the Duty Fire Officer, and by the use of non-standard R/T procedures.

3. CONCLUSIONS

3.1 The crew were properly licensed and qualified to conduct the flight.

3.2 The aircraft was properly certificated and had been correctly maintained.

3.3 The aircraft was being operated within its prescribed limitations.

3.4 There was no pre-crash failure of the aircraft or its systems.

3.5 The ALIA Flight Operations Manual provides specific guidance related to thunderstorm and wind shear encounters.

3.6 The flight crew did not take proper account of the Flight Operations Manual guidance relative to thunderstorm and wind shear encounters.

3.7 The aircraft was forced to the ground in a downburst the effects of which exceeded the climb performance of the aircraft.

3.8 The Airport Fire Service was manned and equipped to International Civil Aviation Organization Category 7 and was adequate for B-727 operations.

3.9 The main strength of the airfield Fire Service did not arrive at the scene of the accident until approximately eleven minutes after the fire had started.

3.10 Cause

The cause of the accident was the aircraft's encounter with a downburst associated with a thunderstorm, the effects of which exceeded the performance capability of the aircraft. The encounter resulted from the flight crew's decision to conduct an approach to land without due regard to the prevailing weather conditions.

ICAO Note: Appendices to this report are not reproduced.

ICAO Ref.: 034/79

No. 4

Boeing 727-31, N840TW, accident near Saginaw, Michigan, United States, on 4 April 1979. Report No. NTSB-AAR-81-8, dated 9 June 1981, released by the National Transportation Safety Board, United States.

SYNOPSIS

About 2148 e.s.t., on April 4, 1979, a Trans World Airlines, Inc., Boeing 727, operating as Flight 841, entered an uncontrolled maneuver at 39,000 feet pressure altitude near Saginaw, Michigan. The aircraft descended to about 5,000 feet in about 63 seconds before the flightcrew regained control. About 2231, the flightcrew made an emergency landing at Metropolitan Airport, Detroit, Michigan. Of the 89 persons aboard, 8 passengers received minor injuries. The aircraft was damaged substantially.

The flight was cruising in visual flight conditions at night at 39,000 feet when the uncontrolled maneuver began; there was no turbulence. There was a cloud layer near 20,000 feet, and, at 2155, the reported weather at Saginaw was 500-foot overcast with 3 miles visibility in light snow; small breaks were reported in the overcast.

Analysis of the evidence indicated that the uncontrolled maneuver began about 2147:47 with isolation of the aircraft's No. 7 leading edge slat (on its right wing) in the extended or partially extended position. During the preceding 14 seconds, the aircraft had rolled slowly to the right to about 35° of right bank and was returned to near wings level flight. Thereafter, the aircraft rolled again to about 35° of right bank in about 4 seconds. About 2147:51, the right roll was stopped near 35° of bank for a few seconds. At that time, the aircraft reached a condition wherein mach number, angle of attack, and sideslip combined to reduce the aircraft's lateral control margin to zero or less, and the aircraft continued to roll to the right in a descending spiral. During the following 33 seconds, the aircraft completed 360° of roll while descending to about 21,000 feet. The aircraft entered a second roll to the right during which the No. 7 slat was torn from the aircraft. Control of the aircraft was regained about 2148:58 at an altitude of about 8,000 feet.

The Safety Board determines that the probable cause of this accident was the isolation of the No. 7 leading edge slat in the fully or partially extended position after an extension of the Nos. 2, 3, 6, and 7 leading edge slats and the subsequent retraction of the Nos. 2, 3, and 6 slats, and the captain's untimely flight control inputs to counter the roll resulting from the slat asymmetry. Contributing to the cause was a preexisting misalignment of the No. 7 slat which, when combined with the cruise condition airloads, precluded retraction of that slat. After eliminating all probable individual or combined mechanical failures, or malfunctions which could lead to slat extension, the Safety Board determined that the extension of the slats was the result of the flightcrew's manipulation of the flap/slat controls. Contributing to the captain's untimely use of the flight controls was distraction due probably to his efforts to rectify the source of the control problem.

1. FACTUAL INFORMATION

1.1 History of the Flight

On April 4, 1979, Trans World Airlines Flight 841, a Boeing 727-31 (N840TW), operated as a scheduled passenger flight from John F. Kennedy International Airport, New York (JFK), to Minneapolis-St. Paul International Airport, Minnesota. After a delay of about 45 minutes due to traffic congestion, Flight 841 departed JFK with 82 passengers and 7 crewmembers aboard at 2025. 1/

About 2054, Flight 841 reached flight level (FL) 350, 2/ to which it had been cleared. At 2124, the flight called Toronto Center and asked for any report on winds at FL 310 or FL 390. The Toronto Center controller replied that he had no reports from other flights. Flight 841 stated that it was encountering a headwind of 100 knots or more, and about 2125, the flight requested clearance to FL 390. The flight was cleared to FL 390, and at 2138:44, it reported reaching FL 390. The captain stated that he climbed the aircraft at 0.80 mach, leveled the aircraft at 39,000 feet at that speed, and engaged the autopilot in the Altitude Hold mode.

According to the fuel and flight data log, at 2140 the second officer estimated that the aircraft's gross weight was 131,700 pounds. According to the flightcrew, the takeoff, climb, and en route portions of the flight were uneventful and no problems occurred until about 9 minutes after the aircraft reached FL 390. The captain stated that the flight was in visual flight conditions at FL 390 and that there was no turbulence. The flight was cruising at about 252 KIAS with all systems indicating normal operation. There were no warning lights visible, and no changes were made to the aircraft's configuration.

The captain stated that he was flying the aircraft on autopilot with the Altitude-Hold mode selected. While he was sorting maps or charts, which were located in his flight bag on the left cockpit floor, he felt a buzzing sensation. Within 2 or 3 seconds, the buzzing became a light buffet, and he looked at the flight instruments. He noticed that the autopilot was commanding a turn to the left with the control wheel displaced accordingly, but he noticed that the attitude director indicator (ADI) showed the aircraft in a 20° to 30° bank to the right. The ADI showed that the aircraft was continuing to bank to the right at a slightly faster than normal rate of roll, so he disconnected the autopilot and applied more left aileron control to stop the roll.

According to the captain, the aircraft continued to roll to the right in spite of nearly full left aileron control, so he applied left rudder control in addition to the aileron control. He stated that in spite of the almost full deflection of the left aileron and full displacement of the left rudder pedal, the aircraft continued to roll to the right. He believed that the aircraft was going to roll inverted so he retarded the throttles to the flight idle position, and he stated "we're going over," or something to that effect. The aircraft rolled completely and entered a second roll with the nose down.

The captain asked the first officer to "get them up," meaning that he wanted the first officer to extend the speed brakes. The first officer stated that he was not aware of the buffeting or the aircraft's attitude because he was in the process of calculating the aircraft's groundspeed; therefore, he did not understand the captain's

1/ Unless otherwise noted, all times herein are eastern standard, based on the 24-hour clock.

2/ A level of constant atmospheric pressure related to a reference datum of 29.92 inchs of mercury; for example, FL 350 represents a barometric altimeter indication of 35,000 feet.

command. The flight engineer was aware of the buffeting but was facing his panel and was not aware initially of the aircraft's attitude except that it seemed to be in a right descending turn. The captain stated that when the first officer did not react to his command, he moved the speed brake lever to the deployed position.

After detecting no reaction to the speed brake extension, the captain moved the control handle to the retract position and back to the extend position. Meanwhile, the indicated airspeed needle was moving rapidly toward its limit and he could see only "black" on the ADI and bright areas in the windshield which he perceived to be the lights of towns shining through the undercast. The altimeter indicated such a rapid descent that it was difficult to read. However, he estimated that the aircraft was near 15,000 feet and descending rapidly when he commanded extension of the landing gear. The first officer immediately moved the gear handle to the "extend" position, and the flightcrew heard a very loud sound similar to the sound of an explosion.

The captain stated that he applied full left aileron and full left rudder throughout the descent but the aircraft continued to roll to the right. Simultaneous with the gear extension, he relaxed some of the back pressure on the control column and some of the pressure on the aileron and rudder controls. The airspeed began to slow, and he was able to roll the aircraft to a near wings-level attitude and to stop the aircraft's descent, after which the aircraft pitched upward into a 30° to 50° climb. He saw the moon in the windscreen and used it as a visual reference to maneuver the aircraft. The airspeed slowed rapidly, and with guidance from the first and second officers, he leveled the aircraft near 13,000 feet.

After regaining control of the aircraft, the flightcrew noticed a warning light announcing the failure of the "A" hydraulic system and a warning flag indicating that the lower yaw damper was inoperative. The captain decided to land the aircraft at Metropolitan Airport, Detroit, Michigan. He instructed the first officer and flight engineer to perform emergency checklist procedures and to notify the flight attendants to prepare the passengers for an emergency landing.

The captain stated that when the landing flaps were extended during the approach by means of the alternate extension system, the aircraft rolled sharply to the left. Therefore, he ordered the flaps retracted and planned for a landing without flaps. The two main landing gear indicators showed unsafe landing gear conditions, so the captain made a low altitude pass down the runway for a check of the landing gear. Control tower and crash rescue personnel reported that all three landing gears appeared to be extended. About 2231, the captain landed the aircraft on runway 3 without incident.

The accident occurred at night (about 2148) near latitude 43°39'N and longitude 84°05'W.

1.2 Injuries to Persons

Injuries	Crew	Passengers	Others	Total
Fatal	0	0	0	0
Serious	0	0	0	0
Minor/None	7	82	0	89

1.3 Damage to Aircraft

The aircraft was damaged substantially; it was repaired and returned to service in late May 1979.

1.4 Other Damage

None.

1.5 Personnel Information

The flightcrew was qualified and certificated for the flight and had received the training required by regulation.

The flightcrew had reported for duty in Los Angeles, California, on April 3, 1979, about 1130. They had flown a series of flights that terminated in Columbus, Ohio, about 2205. On the day of the accident, the flightcrew reported for duty about 1345 and flew to New York, New York, with an en route stop at Philadelphia, Pennsylvania. They arrived at JFK about 1720. All members of the flightcrew stated that they felt no fatigue on April 4.

The captain of Flight 841 first qualified as a captain on B-727 aircraft on February 3, 1969. Later, he had flown as a first officer or captain on various aircraft. Most recently, he had flown as a first officer on B-747 aircraft from November 1977 to December 1978. From late December 1978 until March 11, 1979, the captain was on medical leave, recovering from a broken ankle. On March 15 and 16, 1979, he took a ground school refresher course in the B-727, and on March 19 and 20, he flew the B-727 simulator for 4 hours. On March 21, he received a simulator check, and he made three landings in the B-727 aircraft. On March 28, the captain successfully completed a line check in the B-727 which lasted 5 hours, 21 minutes. On April 3, he began his first line trip since returning to duty. During the 90-day period preceding the accident, the captain flew 21 hours 50 minutes, all in the B-727.

1.6 Aircraft Information

N840TW was owned and operated by Trans World Airlines, Inc. (TWA), and was certificated and maintained in accordance with current regulations. It was purchased from The Boeing Company on July 13, 1965. N840TW had acquired about 35,412 hours in service.

The aircraft received a "C" check on March 1, 1979, and it had been flown 230 hours 13 minutes since that check. Maintenance records indicated that during the "C" check, suspected hydraulic leaks in the No. 8 spoiler actuator, No. 4 and No. 5 leading in edge flap actuators, and No. 6 and No. 7 leading edge slat actuators were either invalidated or were repaired. The No. 7 leading edge slat's inboard track fairing was repaired. There were no significant maintenance discrepancies on the aircraft maintenance logs after the "C" check.

The aircraft's planned gross weight for takeoff was 145,095 pounds (lbs) with 36,000 lbs of fuel on board. About 1,500 lbs of fuel were consumed during the delay preceding the takeoff. At the time of the accident, the aircraft's center of gravity was within prescribed limits at 24.1 percent mean aerodynamic chord and the aircraft's gross

weight was about 130,400 lbs. After the aircraft had landed and after the engines were stopped, according to the aircraft's fuel gages, 13,890 lbs of fuel were on board: 4,580 lbs in the No. 1 tank; 4,710 lbs in the No. 2 tank; and 4,600 lbs in the No. 3 tank.

1.7 Meteorological Information

At 1900 on April 4, 1979, the National Weather Service's (NWS) upper air analysis showed southwesterly winds at the 200-, 250-, and 300-millibar levels 3/ through New York and Michigan. The wind speeds were 100 to 110 knots in eastern New York, 35 to 50 knots in western New York, and 80 to 85 knots in east-central Michigan. In eastern and central Michigan, the air temperature at the 200-millibar level was about -49° C.

The 1800 radiosonde observation at Flint, Michigan, showed temperatures of -48.9° C at 38,000 feet and -53° C at 44,400 feet. Near 39,000 feet, measured winds were from 230° true at 85 knots. The tropopause was near 30,000 feet.

The surface weather observations at the following times and locations were, in part:

Saginaw, Michigan

2155 - Clouds—measured ceiling 500-ft overcast; visibility—3 mi in light snow; wind—350° at 5 kns; remarks—small breaks in the overcast.

Detroit, Michigan

2153 - Clouds—800-ft scattered, measured overcast ceiling at 2,000 ft; visibility—7 mi; wind—310° at 11 kns; remarks--snow ended at 2135.

NWS weather radar observations taken at Detroit at 2130 and 2230 showed that no precipitation echoes existed within 250 miles of Detroit.

Weather reports submitted by pilots were, in part, as follows:

2019 - Peck VOR, Michigan, FL 310—sky clear, no turbulence, temperature -45° C, winds 270° at 80 kns.

2026 - Flint, Michigan, FL 350—sky clear, no turbulence, temperature -48° C, winds 240° at 100 kns.

According to U.S. Naval Observatory astronomical data, on April 4, 1979, at 2150, at latitude 43°39'N and longitude 84°05'W, a half moon was visible at an azimuth of 242° from true north and at an elevation of 48° above the horizon.

1.8 Aids to Navigation

Navigational aids were not a factor in this accident.

1.9 Communications

There were no problems with communications.

3/ Pressure levels corresponding approximately to pressure altitudes of 39,000, 34,000, and 30,000 feet, respectively.

1.10 Aerodrome and Ground Facilities

There were no problems with the aerodrome or ground facilities.

1.11 Flight Recorders

The aircraft was equipped with a Lockheed Aircraft Services Model 109-D flight data recorder (FDR), serial No. 219. The recording foil was not damaged and all four flight parameters were clear and active. (See appendix C.) There was no evidence of malfunction except at one point in the heading trace where the heading stylus moved in a direction opposite to normal movement of the recording foil while the aircraft was in a turn and was being subjected to high vertical acceleration forces. Further examination of the heading trace disclosed that this abnormality occurred again when the aircraft was turning off the runway at Detroit. A detailed examination of the recorder heading trace mechanism disclosed no explanation for these abnormalities. However, according to the manufacturer, the backward movement of the heading stylus (apparent time shifts) were caused by worn mechanisms in the FDR.

The aircraft was equipped with a Fairchild Industries Model A-100 cockpit voice recorder (CVR), serial No. 829. The CVR was not damaged; however, 21 minutes of the 30-minute tape were blank. The remaining 9 minutes of tape were of good fidelity, but they pertained only to flightcrew conversations after the aircraft was on the ground at Detroit.

Tests of the CVR in the aircraft revealed no discrepancies in the CVR's electrical and recording systems. The CVR tape can be erased by means of the bulk-erase feature on the CVR control panel located in the cockpit. This feature can be activated only after the aircraft is on the ground with its parking brake engaged. In a deposition taken by the Safety Board, the captain stated that he usually activates the bulk-erase feature on the CVR at the conclusion of each flight to preclude inappropriate use of recorded conversations. However, in this instance, he could not recall having done so. The first and second officers both stated that they did not erase the tape nor did they see the captain activate the erase button on the CVR control panel.

1.12 Wreckage and Impact Information

The No. 7 leading edge slat on the right wing was missing. The slat tracks remained on the aircraft; the outboard track was twisted and bent rearward about midspan, and the inboard track was bent rearward near the aft end of the track. The slat actuator cylinder was broken about 1 1/2 inches forward of its trunnion; the aft portion of the cylinder remained attached to the wing. The forward end of the actuator cylinder, the actuator piston, and the piston rod were missing. The 5/16-inch bolts that attach the slat to its track were sheared. The inboard fairing-adjustment T-bolt was broken, and the threaded portion of the bolt and two adjusting nuts were missing. The inboard slat hook showed no evidence of engagement with the hook stop - the chromate paint was intact while the paint on the outboard hook was partially worn.

The skin of the lower surface of the wing aft of the No. 7 slat actuator was scraped. An 8- to 10-inch portion of the outboard aileron balance tab was missing at the end of the scrape mark. The balance tab actuator lugs had separated, and the hinge support fitting between the lugs had sheared.

The following components were removed from the airplane and were functionally tested: (1) Kollsman integrated flight instrument system, (2) captain's and first officer's airspeed and mach indicators, (3) yaw dampers, (4) autopilot control panel and pitch roll channels, (5) air data sensor, and (6) both instantaneous vertical speed indicators. All components, except one airspeed indicator, functioned within specified tolerances; the airspeed indicator was about 4 knots out of tolerance in the 240- to 260-knot speed range.

The No. 4 and No. 10 spoiler actuators were tested and they functioned satisfactorily. The remaining portion of the No. 7 slat actuator, including the switch mechanism, one lockring, and the shuttle valve, was examined. A production piston assembly could not be inserted into the broken end of the cylinder bore, but after removal of the end cap, it could be inserted into the opposite end of the bore and into the normal retracted position. However, the piston could not be moved past the retracted position through the broken end of the bore. Dimensional analysis of the bore disclosed that it was distorted near the broken end. The actuator bore was not scored, scraped, or marked.

Further tests at the aircraft manufacturer's facility indicated that when 3,000 psi of hydraulic pressure was applied to the extend face of the production piston, the piston would not move out of the broken end of the actuator bore; instead, the hydraulic fluid leaked between the bore distortions and the piston seal. A mechanical force of 1,025 pounds was required to force the piston out of the broken end of the actuator bore; the actuator bore was gouged and scraped by the piston as it moved through the bore.

1.13 Medical and Pathological Information

The flightcrew was not examined medically.

Of the five passengers who immediately reported injuries, two passengers were taken by ambulance to a local hospital where they were treated and released. Three passengers reported pains in their chests, necks, and backs, but they refused medical treatment. One passenger's knee was bruised and bleeding, and her ankle was swollen. The passengers' injuries consisted primarily of strains and bruises. All five passengers flew to Minneapolis-St. Paul on another flight which departed Detroit about 0245 on April 5, 1979. Later, three other passengers reported injuries, but only one was hospitalized for severe muscle strain of the back and neck and a vertigo/balance problem.

1.14 Fire

There was no fire.

1.15 Survival Aspects

This was a survivable accident. The injury causing mechanism was the variable but comparatively high in-flight load factor -- maximum of about 6.0 g's -- and its duration. The high g's forced the occupants' heads and upper extremities toward the floor of the cabin and caused the muscle strains of the neck and back. Passengers who were standing when the maneuver began were forced to the floor and, in the process, contacted objects that caused bruises and cuts.

1.16 Tests and Research

The right outboard aileron actuator hinge fitting bolt was broken. With the aileron in the locked-out position, there was free movement of 1 inch up and 3/32 inch down at the trailing edge of the aileron. The nut end of the bolt remained in the structure. A metallurgical examination of the bolt indicated that it had failed predominantly in fatigue.

The No. 10 flight spoiler panel, except for a portion containing the two inboard hinges, was missing. The right inboard trailing edge flap track attachment bolts were sheared and the carriage was damaged. The canoe-shaped fairing for the track was missing.

The No. 7 leading edge slat, which had broken into two pieces, and the right outboard trailing edge flap track canoe-shaped fairing were found about 7 miles north of Saginaw, Michigan, at latitude 43°39'N and longitude 84°05'W. A large portion of the No. 10 spoiler panel was found about 3/4 mile south of these components. The forward portion of the No. 7 slat actuator cylinder, the actuator piston, and the piston rod were not found. The piston rod-end bearing remained attached to the slat; the rod had fractured in overload about 2 inches aft of the center of the bearing.

A metallurgical examination of the No. 7 slat inboard T-bolt indicated that the cross section of the bolt had fatigue fracture characteristics. There was considerable smearing of the fracture face.

Both main gear landing doors and their operating mechanisms were damaged extensively and a hydraulic line was ruptured. The sidebrace and actuator support beam on the right gear were broken; the support beam for the left gear was intact. The uplock for the left gear was bent. The secondary wing skin panels above both actuator support beams were buckled upward.

The No. 4 flight spoiler was torn around its actuator attachment point. Fuel was leaking around several structural fasteners in the left wing. The aft fairing on the left outboard trailing edge flap jackscrew was broken and the forward fairing was missing. The left outboard aileron balance tab hinge fitting was broken; in the locked-out position, there was no appreciable free movement of the aileron.

Slight tension-field wrinkles had formed in the fuselage skin fore and aft of the wing attachment areas. The nose gear door was damaged. Both inboard flap jackscrew fairings were loose and both transmission housings were broken. Two blowout panels on the bottom of the No. 3 engine support strut were missing.

Many passenger oxygen masks were hanging from their overhead compartments. A passenger service unit was loosened from its moorings and an interior window was cracked.

The "A" hydraulic system reservoir contained 2 quarts of fluid. Following repair of the hydraulic line in the right wheel well and plugging of the No. 7 slat actuator lines, the reservoir was serviced and the flight controls and speed brakes were checked; they functioned properly. Except for the No. 7 leading edge slat, the leading edge slats and flaps, trailing edge flaps, and their indicator lights functioned properly on both the normal and alternate flap systems. The inboard trailing edge flaps could not be tested because of the damage to the right inboard trailing edge flap. The stall warning and overspeed warning systems functioned properly.

1.16.1 Boeing Company Tests

In 1975, The Boeing Company conducted flight and wind tunnel tests to determine the effects of asymmetric extension of wing leading edge slats on the control characteristics of the B-727 while in cruise flight conditions. Because of reports of slat actuator lockring failures, these tests were conducted to evaluate control characteristics associated with an unscheduled extension of a single leading edge slat. The wind tunnel tests involved slat extensions from 0.4 to 0.95 mach; because of adverse buffeting, 0.8 mach was the highest speed tested in flight. From these tests, it was determined that the extension of either the No. 2 or the No. 7 leading edge slat caused the most adverse control characteristics, but with a significant amount of lateral control applied, the aircraft was controllable at altitudes and speeds of up to and including 35,000 feet and 0.80 mach. As the result of these tests, The Boeing Company issued Operations Manual Bulletin (OMB) 75-7 in August 1975, to provide flightcrews with operational information for the event of an actual or suspected leading edge slat actuator lockring malfunction.

1.16.2 Flight Simulator Tests

At the request of the Safety Board, The Boeing Company programed a fixed-base engineering flight simulator with B-727-200 4/ aerodynamic and control data and the data obtained from the 1975 flight and wind tunnel tests. Also, the simulator was programed with Flight 841's gross weight and center of gravity conditions and the pertinent meteorological data associated with its flight. A total of 118 trials were conducted in the flight simulator to identify the condition that precipitated the aircraft's upset and to duplicate and evaluate its maneuver.

The effects of spurious signals to the autopilot and yaw damper were explored. In all cases, the simulator pilots were able to overcome the effects with no difficulties. There was no correlation between the simulator traces of these maneuvers and Flight 841's FDR traces.

The simulator tests did not take into consideration the effects of freeplay in the right out-board aileron or possible distortion of the No. 7 leading edge slat. Also, the simulations did not include lateral control requirements due to sideslip. However, the effect of 1 3/32 inches of free play in the right outboard aileron was explored. About 13° of control wheel deflection to the left was needed to counter the effect; this deflection was within the autopilot's authority of about 38° of control wheel deflection. The control wheel must be deflected about 80° to obtain full deflection of the lateral controls.

Extensions of the No. 7 leading edge slat while in level flight at 39,000 feet and at 0.80 mach consistently generated rolls to the right. Additionally, the autopilot countered the right rolls to the full extent of its authority within 2 seconds of slat extension, and the yaw damper countered the resulting sideslip with rudder displacement to the left. The initial roll rate with the autopilot engaged was about 7° per second to the right. Despite the countercontrols applied by the autopilot, the simulated aircraft continued to roll to the right. When the pilots disconnected the autopilot at 30° of right bank and applied additional left aileron control, the simulated aircraft returned to

4/ According to the Boeing Company, the control systems and aerodynamic characteristics of the B-727-200 are virtually identical to the B-727-100. The only aerodynamic differences relate to a 10-foot longer fuselage on the B-727-200.

wings-level flight which could be maintained with about 46° of control wheel displacement to the left. Under conditions of the simulations, the right roll could be stopped and the aircraft returned to wings-level flight provided corrective measures were taken before the roll progressed beyond about 117° of right bank. Beyond that bank angle, increases in angle of attack and mach produced an imbalance of rolling moments that could not be overcome with lateral and rudder controls, and the aircraft became uncontrollable.

The simulator tests produced two flight maneuver situations in which the recorded time histories of indicated airspeed, altitude, and normal load factors most closely approximated those recorded by Flight 841's FDR. The recorded time history of the simulator's heading trace differed from the FDR heading trace because the simulator's heading reference system was not subject to the gimbal errors associated with the aircraft's heading gyro.

In both maneuvering situations, the entry and the pilots' actions were similar; the latter differed only in the amount of controls applied and the length of delay in control application following slat extension. The simulator was placed at 39,000 feet at 0.80 mach with the autopilot engaged. After extension of the No. 7 leading edge slat and a short delay, the pilots disconnected the autopilot and applied left aileron control, followed by left rudder deflection. The pilots retarded the throttles to flight idle and extended the speed brakes. Near 15,000 feet, they extended the landing gear and the slat was retracted (simulating the loss of the slat) and recovery to level flight was completed. Since load factors could not be simulated, the pilots attempted to duplicate Flight 841's longitudinal control deflections and g-trace by referring to a g-meter. The stabilizer trim was not moved from the cruise trim (zero stick-force) position.

The flight simulator traces showed that the simulated aircraft could be returned to wings-level flight with relatively little loss of altitude provided corrective action was begun before the roll and airspeed were allowed to increase excessively. In the simulations, the pilot could delay reaction for about 16 seconds and regain control with an altitude loss of about 6,000 feet. However, when the pilot delayed corrective action for 17 seconds or more, a maneuver was entered that approximated Flight 841's airspeed, altitude, and g-traces. In this maneuver, the aircraft continued throughout the descent to roll to the right, in spite of full left aileron and rudder, until the slat was retracted to simulate its loss from the aircraft.

The second maneuvering situation which produced good correlation between the simulator traces and Flight 841's FDR traces for airspeed, altitude, and g's involved two roll reversals. About 12 seconds after slat extension and at a bank angle of about 60° to the right, the pilot disconnected the autopilot and rapidly applied left aileron and left rudder. In the following 12 seconds, the simulated aircraft rolled 285° to the left, about 45° beyond the inverted position, and then it reversed and rolled to the right. By that time, the simulated aircraft's nose had dropped to 40° below its cruise attitude and its airspeed had increased to 0.86 mach. After the roll reversal to the right, the simulated aircraft continued throughout the descent to roll to the right, in spite of full left aileron and rudder, until the slat was retracted.

In all simulator tests, during the spiraling descents, lateral control was regained after the No. 7 leading edge slat was retracted to simulate its loss from the aircraft. Several traces from the simulator tests indicated that, while the autopilot was deflecting the control wheel to the left and the aircraft was rolling to the right, fully qualified B-727 pilots disconnected the autopilot, centered the control wheel, and then

rotated the control wheel to the left to counter the right roll. This had the effect of momentarily increasing the roll rate to the right.

1.16.3 Heading Gyro Tests

In tests in May 1979 to determine the effects of pitch and roll angles on heading gyro performance, the heading gyro from the aircraft was removed and mounted on a movable platform. The heading angle indicated by the gyro was displayed on a digital readout of the heading synchro and was recorded. The system was stabilized on Flight 841's cruise heading and attitude which immediately preceded the control problem. The heading gyro was then progressively rotated to the heading, pitch, and roll angles which were measured in the simulator tests that most nearly duplicated the aircraft's FDR traces of airspeed, altitude, and g's.

Comparison of the simulator heading traces with those obtained from the heading gyro tests showed some correlation for the simulator maneuver which developed when the simulator pilot began his corrective action at 60° of right bank and about 12 seconds after slat extension. This was the simulator maneuver which involved double roll reversals; however, about 36 seconds after slat extension, significant disagreement between the heading traces occurred.

In May 1980, additional tests were conducted to determine the effect of roll, pitch, and heading angles changes and rates of change on the performance of a B-727 heading gyro. By determining these effects, it was believed that a better understanding and interpretation could be achieved of the accident aircraft's motions as reflected in its FDR heading trace.

The heading gyro was mounted in a standard bracket and attached to a tilt/turn table. The gyro was connected to a compass coupler and flux gate valve to simulate a complete B-727 compass system, and a Lockheed Air Service Model F109-D FDR was used to record the heading information developed from rotation of the gyro through various roll, pitch, and heading angles. Several tests involved the introduction of roll, pitch, and heading angles that were derived from the two flight simulation maneuvers that most closely approximated the time histories of airspeed, altitude, and g's recorded on Flight 841's FDR.

The heading traces recorded on the test FDR shifted backward in time in a manner similar to the backward shift of the accident aircraft heading trace which occurred about 2148:01. These time shifts were found to occur at specific bank angles, which made them predictable, and corrections were made to remove the time shifts from the accident aircraft's heading trace and from the test traces. According to the FDR manufacturer, these time shifts were caused by worn recorder mechanisms.

These tests established that for changes in roll angle alone, the magnitude and direction of the heading gyro gimbal errors were the same whether the gyro was rolled to the right or to the left, and the heading trace generated on the test FDR was essentially the same regardless of the direction of roll. Also, the gimbal error was a repeatable function of roll attitude and was not affected by roll rate.

With regard to roll, pitch, and heading angle changes associated with flight simulation data for a continuous right roll entry into the dive, comparison of the test heading trace with the accident heading trace showed poor agreement with the abrupt

heading change to the right that occured in the latter trace about 2147:45. However, the comparison showed good agreement with the first large heading excursion to the left that began about 2147:54. Comparison of the test heading trace generated from simulator data for the dive entry which involved two roll reversals showed good agreement with the abrupt heading change to the right that occurred about 2147:45 and fair agreement with the subsequent large heading excursion to the left.

Heading traces based on the above simulation data for roll, pitch, and heading angle changes were calculated from the gimbal error equation and were compared with the accident heading trace. The calculated heading trace for the continuous right roll entry data showed poor agreement with the abrupt heading change to the right that occurred about 2147:45 but showed good agreement with the subsequent large heading excursion to the left. Conversely, the calculated trace for the dive entry which involved two roll reversals showed good agreement with the abrupt heading change to the right but poor agreement with the subsequent large excursion to the left. The calculated trace contained an uncharacteristic inflection when the roll reversed from left to right.

1.16.4 Flight Tests

On October 2, 1980, at the request of the Safety Board, the aircraft manufacturer conducted flight tests in an instrumented B-727-100. A Safety Board aerospace engineer assisted in developing the tests and was aboard the aircraft as an observer during the tests. The purposes of the tests were to record data that could be compared with Flight 841's FDR data, to test aircraft and configuration changes that might have occurred to Flight 841 before its rapid descent, and to obtain data on lateral control effectiveness.

The aircraft was equipped with a Lockheed Aircraft Services 109-D FDR and accelerometer of the same types that were aboard the accident aircraft. Additionally, the aircraft was equipped with special flight instrumentation, a Sperry Flight Systems SP150, MB-5 autopilot, and recorders that recorded substantially more parameters with higher accuracy and fidelity than the FDR. All tests were flown as close as practical to the conditions recorded on Flight 841's FDR just before its g-trace began oscillation. The tests involving configuration changes in level cruise flight were flown for about 1 minute each, following completion of the configuration change, to determine the effects on aircraft performance.

The significant results of these tests were as follows:

- With the autopilot engaged in Altitude Hold and Manual Mode, when the trailing edge flaps were extended to 2° without extension of the leading edge slats, there were no discernible changes in altitude, g, or heading on the FDR. Airspeed decreased very slightly and slowly.

- With the autopilot engaged in Altitude Hold and Manual Mode, when the trailing edge flaps were extended to 5° without extension of the leading edge slats, the altitude increased slightly and the airspeed decreased slowly a few knots. Normal acceleration initially increased slightly, then decreased slightly, and then returned to 1.0 g. When the flaps were retracted to 2°, the airspeed increased slowly about 1 knot.

- With 2° of trailing edge flaps extended and autopilot engaged (Altitude Hold and Manual Mode), when the Nos. 2, 3, 6, and 7 slats were extended, the airspeed decreased at a rate of 0.50 kn/sec., and the altitude remained constant. The g-trace showed an initial increase followed by a periodic undamped oscillation. Moderate buffet (as described by an engineer in the midsection of the aircraft) was indicated on the flight test FDR by g-trace oscillations of ± 0.05 g at a frequency of about 1.0 cycle per second.

- As drawn on a graph, a comparison of the flight test FDR traces with Flight 841's FDR traces suggests that the only similarities are in the initial rate of airspeed decrease and the amplitude of the g-trace oscillations following first excitation of the g-traces. (See appendix F.) However, 200-power magnification of the g-trace on the FDR foils from the flight test aircraft and the accident aircraft showed that the g-trace on the flight test aircraft oscillated about 6 cycles per second when the Nos. 2, 3, 6, and 7 leading edge slats were extended and that the g-trace on the accident aircraft oscillated about 6 cycles per second beginning about 2147:34. The frequency response of the acceleration channel on both FDR's was tested. These tests showed that the accident aircraft's FDR was slightly more responsive and that the frequency response limit of both FDR's was about 6 cycles per second.

  Further, the periodic undamped oscillation of the g-trace from the test aircraft was a characteristic of its autopilot, which was different from the autopilot in the accident aircraft (SP50, MB-3). The primary differences between the two autopilots involve the addition of an altitude rate feedback loop and a higher gain in the altitude feedback loop on the SP150. Additionally, it was determined that during the flight tests higher than normal altitude and altitude rate feedback loop gains were present because a test switch in the digital air data computer (DADC) used by maintenance personnel to functionally test the system had been left in the test (HOLD) position. During simulations of the flight test conditions, the SP150 autopilot (with the DADC test switch in the test position) altitude and altitude rate feedback loops generated nose-up commands to the elevator which partially canceled the nose-down commands to the elevator from the pitch attitude and pitch rate loops. This resulted in a substantial reduction and lag in nose-down elevator response which very nearly duplicated the elevator response recorded during the flight test. Because the SP50, MB-3, autopilot does not have an altitude rate feedback loop and has a reduced gain in the altitude feedback loop, it would have commanded greater nose-down elevator deflections much more rapidly under the same conditions. The effects of earlier and larger nose-down elevator deflections during the flight tests would have been a substantial reduction in the aircraft's initial nose-up pitch attitude and g increase and no subsequent undamped oscillation. Consequently, the flight test aircraft's g-trace would have matched the accident aircraft's g-trace much more closely had the test switch in the DADC been in its proper position. This also was established by simulations of elevator response with the test switch in its proper position.

- While in a normal cruise configuration and with the autopilot (Altitude Hold and Manual Mode) maintaining the aircraft's heading, the pilot deflected the rudder fully to the right to place the aircraft in a steady left sideslip for the purpose of developing a maximum right yawing moment. When the autopilot was disconnected while the aircraft was in this condition, the control wheel abruptly centered and the aircraft slowly yawed, rolled, and turned to the right.

1.17 Additional Information

1.17.1 B-727 Flap System

The B-727 has eight individual leading edge slats and four individual leading edge flaps. The slats are mounted on the outboard positions of the wings and are numbered 1 through 4 on the left wing and 5 through 8 on the right wing. The flaps are on the inboard portion of the wings. (See figure 1.) The leading edge slats and flaps are high lift devices that are extended for takeoff and landing. (See figure 2.) According to the TWA Flight Handbook, the maximum speed and altitude with the leading edge devices extended are 240 knots and 25,000 feet, respectively, and the maximum speed for extension and retraction is 230 knots.

Each leading edge device is actuated by a single hydraulic actuator. These actuators are normally supplied by "A" system hydraulic pressure and are normally controlled by the flap handle in the cockpit. With the loss of "A" system pressure, the leading edge devices can be extended by an alternate flap system; once extended by the alternate system, the leading edge devices cannot be retracted until "A" system pressure is restored. When extended by the alternate system, all leading edge slats and flaps extend randomly; full extension of all devices takes about 40 seconds.

The leading edge devices normally extend and retract in conjunction with the trailing edge flaps. The normal schedule is for extension of the Nos. 2, 3, 6, and 7 leading edge slats in conjunction with the selection of 2° of trailing edge flaps and extension of the No. 1, 4, 5, and 8 slats and the six leading edge flaps in conjunction with the selection of 5° of trailing edge flaps. The devices are retracted in the same groups in the reverse order of extension. Actual extension and retraction are initiated by the outboard trailing edge flap followup system; that is, slat extension is not initiated until the trailing edge flaps approach the 2° and 5° positions and slat retraction occurs when the trailing edge flaps retract from the 5° and 2° positions. Normal extension and retraction times for the leading edge devices is about 6 seconds.

All leading edge slats are held in the extended and retracted positions by hydraulic pressure and by mechanical locking devices in each actuator. (See figure 3.) The locking devices are held in the locked position by springs and hydraulic pressure and are unlocked by hydraulic pressure. A switch on each actuator is connected to the cockpit light displays. One display, on the pilots' instrument panel, provides an amber (in transist) light when an acutator is unlocked, a green light when all actuators are extended and locked, or no light when the actuators are retracted and locked. The other display, aft of the flight engineer's panel, provides a display of the condition of each individual leading edge device. When activated, this display will show whether an individual leading edge device is extended and locked (green light), retracted and locked (no light), or unlocked (amber light).

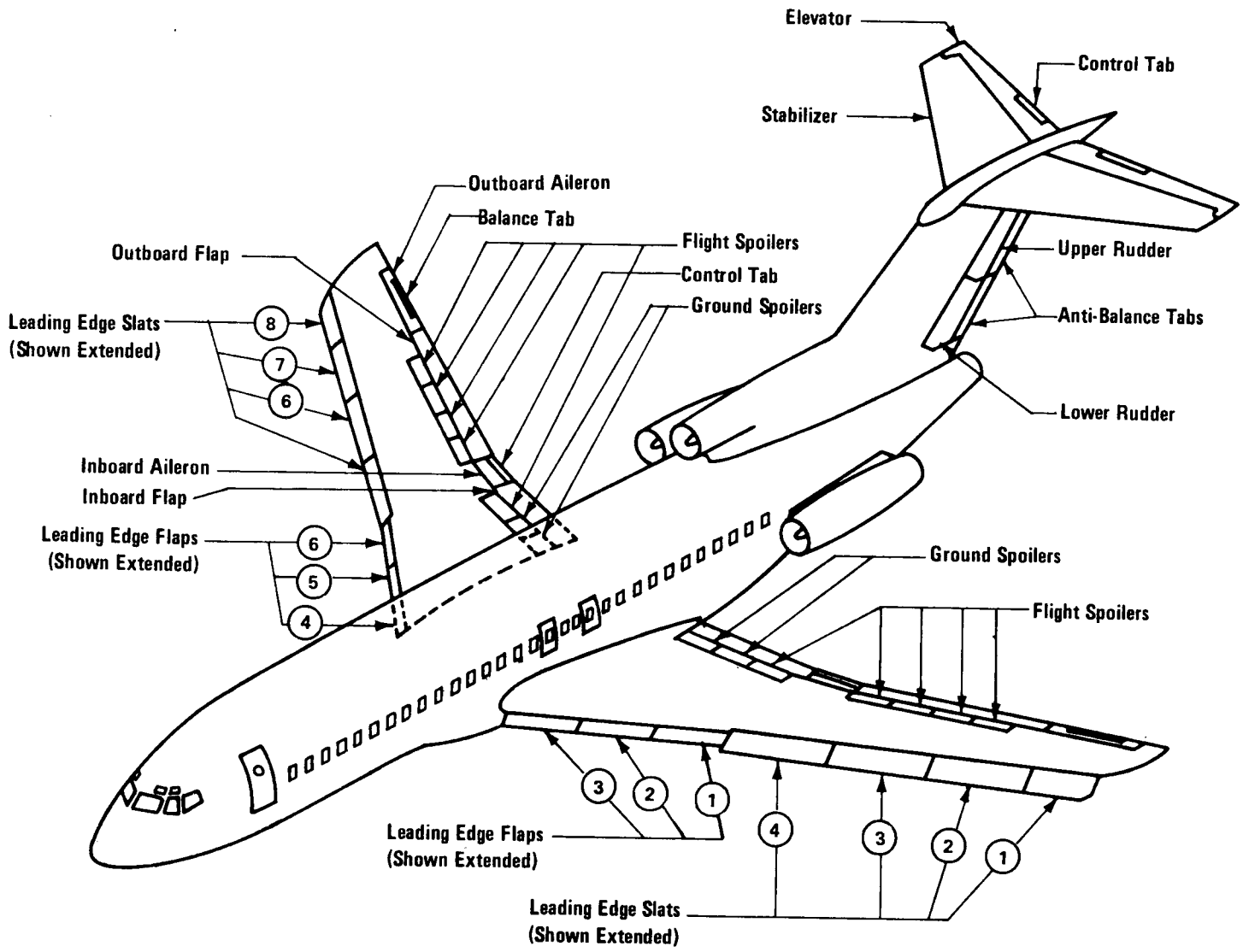


Figure 1.—B-727 flight control system surfaces.

1.17.2 History of B-727 Leading Edge Slat Problems

According to FAA service difficulty reports (SDRs), from the beginning of 1970 through the end of 1973, seven cases of a single leading edge slat extension and separation on B-727's during flight were reported without mention of whether the extensions were scheduled or unscheduled. In 1974 and 1975, no unscheduled extensions and separations were reported. In 1976, one unscheduled extension without separation was reported; the slat actuator support fitting had broken. No other unscheduled extensions and separations have been reported.

The SDRs contain numerous reports of a leading edge device failing to extend or retract, but these failures were associated with normal extension or retraction schedules. Also, a number of reports attributed the loss of "A" system pressure to a leaking actuator. No uncontrollable flight situations were associated with any of the above failures to extend or retract.

In 1978, one operator experienced an unintended extension of leading edge devices. While in cruise flight at 25,000 feet and about 350 knots (0.82 mach), the captain detected an airframe vibration which he attributed to a partially extended trailing edge flap. He attempted to retract the trailing edge flap by using the alternate flap system. However, either the retraction switch was moved inadvertently to the "down" position, rather than the "up" position, or the switch was wired backward. In any event, the leading edge devices were unintentionally extended. The leading edge devices were retracted by turning the alternate flap master switch off; however, the No. 6 and No. 7 leading edge slats on the right wing did not retract. The aircraft began to roll and turn to the right, but the captain returned the aircraft to level flight by using left aileron and rudder. The aircraft was kept upright by about 45° of control wheel deflection to the left and by a significant amount of left rudder. After the captain slowed the aircraft, the slats retracted. An unscheduled but normal landing was made as a precautionary measure. The No. 7 slat and the alternate flap retraction switch were changed, and the aircraft was returned to service.

1.17.3 Aircraft Performance

Correlation of FDR information with air traffic control data showed that, after Flight 841 reported level at FL 390 at 2138:44, the FDR vertical acceleration trace remained steady at 1.0 g. At 2147:34, near the Saginaw, Michigan, VOR, while on a steady heading of 288°, the g-trace began to oscillate at an amplitude of about ±0.05 g and at a frequency of about 2.0 cycles per second. These oscillations continued for 70 seconds with the amplitude increasing to a maximum of about ±0.3 g.

After the vertical acceleration trace began to oscillate, the airspeed began to decrease from 245 knots and 10 seconds later it was 240 knots. Also, the heading trace deviated about 1° during the first 6 seconds of the oscillation; it then moved erratically to a heading of about 298° during the following 13 seconds, including an abrupt change of about 5° to the right in 0.5 second, beginning at 2147:45. During the first 19 seconds of g-trace oscillation, the altitude trace decreased from 39,600 feet 5/ to 39,000 feet. From 2147:53 to 2148:04, the altitude decreased to 37,500 feet and the heading trace moved to 184°. During that period, the airspeed increased to 250 knots and the g-trace increased to about 1.7 g's.

5/ Recorder tolerances at 39,000 feet are ± 700 feet.

From 2148:04 to 2148:28, the heading trace moved to 360°, the altitude decreased to about 19,500 feet, the airspeed increased to about 390 knots, and the g-trace increased to 4.0 g's. During the next 17 seconds, the airpseed increased to its maximum value of 470 knots, the altitude trace decreased to its minimum value of about 5,000 feet, and the heading trace moved to about 310°. Also, the g-trace increased to 5.5 g's.

At 2148:51, after having decreased to about 4.5 g's, the g-trace increased rapidly to 6.0 g's; during the following 7 seconds, the g-trace decreased to 1.0 g. At 2148:58, the airspeed was 280 knots and the altitude was 8,500 feet. During the following 27 seconds, the g-trace varied between 1.0 and 0.3 g, the heading trace moved to about 240°, the altitude increased to about 11,300 feet, and the airspeed decreased to about 160 knots.

According to data in the aircraft flight manual, while in cruise flight at 39,000 feet, a gross weight of 130,000 pounds, and about 245 knots indicated airspeed (mach 0.80), Flight 841's maneuvering and performance margins were about 1.37 g's, 70 knots above 1.0 g low speed stall buffet, and 36 knots below 1.0 g high speed (mach) buffet. In smooth air, the aircraft could have sustained level flight at 43° of bank without entering stall buffet.

The FDR airspeeds recorded during Flight 841's descent from 39,000 to about 5,000 feet were converted to mach numbers by applying position error, compressibility, and density altitude corrections. As the aircraft descended, the initial cruise mach number increased to a maximum of 0.96 mach at 31,800 feet and then decreased to 0.78 mach at 10,000 feet and about 0.70 mach near 5,000 feet.

Wing lift coefficents were calculated from FDR airspeed, altitude, and vertical acceleration values. These coefficients fluctuated throughout the descent. High coefficients of 0.7, 0.7, and 1.08 occurred near 39,000, 21,000, and 7,000 feet, respectively. Low values of 0.28 and 0.51 occurred at 30,000 and 16,000 feet, respectively. These lift coefficients were compared to buffet lift coefficients derived from flight test data. The results indicate that Flight 841 was in a high speed buffet throughout most of the descent and recovery to level flight.

Flight 841's flightpath angles during its descent were calculated from indicated airspeed and rate of descent values. These calculations indicated that the aircraft's flightpath angle decreased from zero at 39,000 feet to 90° vertically downward near 29,000 feet. The angle then decreased to about 30° downward at 24,000 feet and then increased to 40° downward as the aircraft descended through about 18,000 feet. The angle then increased further to about 80° downward at 11,000 feet, decreased to zero during the recovery, and increased to about 55° upward during the pullup following recovery. The angle decreased to about zero near 11,000 feet.

The rolling moment coefficients caused by the extension of the No. 7 leading edge slat were supplied by the aircraft manufacturer from wind tunnel and flight test data developed in 1975. These coefficients were compared to the lateral rolling moments available from aileron and spoiler controls at various mach numbers and altitudes as a function of wing angle of attack. The comparison showed that at Flight 841's cruise altitude of 39,000 feet and at a cruise angle of attack of 4°, the rolling moment available from lateral controls at mach 0.80 exceeded the rolling moment caused by the extended slat by a factor of about 2. At 37,000 feet, mach 0.85, and an angle of attack of 5°, the rolling moments were approximately equal. Further increases in the angle of attack or

mach number during the descent generated rolling moments from the extended slat that exceeded those available from the lateral controls. For instance, at a 5° angle of attack, mach 0.90, and 33,000 feet, the rolling moment generated by the extended slat exceeded the moment available from lateral controls by a factor of about 2.4. When the slat was separated from the aircraft's wing, the rolling moments available from lateral controls were dominant.

The manufacturer later provided data to define the critical lateral control margins of the accident aircraft based on assumptions that the right outboard aileron hinge fitting bolt was broken (allowing the aileron to float at its zero hinge moment position) and that the No. 7 leading edge slat extended suddenly to its most adverse position, including the possibilities of deflected and cocked positions. The data were based on the rolling moment coefficient obtained from a flight test data point (at mach 0.80), rather than a wind tunnel test data point.

These data showed that the adverse incremental rolling moment coefficients produced by a floating right outboard aileron were essentially negligible. Also, the data showed that at angles of attack of less than 5° the most adverse position of the slat was its normal extended position. However, the use of the flight test data point (see figure 4) showed that at mach 0.80 and an angle of attack of 4°, the rolling moment coefficient available from lateral controls exceeded the rolling moment coefficient caused by the extended slat by a factor of 1.26, and, that at mach 0.85 at 35,500 feet, the rolling moment coefficients were approximately equal at an angle of attack of 4.8°. Also, by interpolation, at mach 0.83 and an angle of attack of 6°, the moments were approximately equal. Any further increases in either the mach number or angle of attack caused the rolling moment coefficient produced by the extended slat and a floating aileron to exceed the coefficient available from the lateral controls. From the time histories of Flight 841's mach number and angle of attack, it was determined that the rolling moment equalization point occurred 34.5 seconds after the first oscillation of Flight 841's g-trace, or about 2148:07 e.s.t.

Using the adjusted rolling moment data, a dynamic analysis of Flight 841's maneuver was performed. This analysis showed that 12.5 seconds after development of the roll to the right caused by the extended No. 7 slat, the simulated aircraft reached an attitude such that lateral control could not be regained by full deflection of lateral and yaw controls.

1.17.4 No. 7 Leading Edge Slat Operation

The aircraft manufacturer supplied data from the original design wind tunnel and flight tests to determine the forces acting on the No. 7 leading edge slat actuator rod when the slat is retracted. These data showed that, at an equivalent airspeed of 300 knots, the actuator rod would have a compressive load of about 1,400 pounds on it at mach 0.80. Since Flight 841's equivalent airspeed was about 229 knots (245 knots indicated airspeed) at 39,000 feet, rod loads were calculated for these conditions. Calculations showed that at 1.0 g, compressive loads of 700 pounds would act on the rod with the slat retracted. If the g was reduced to 0.70, the compressive load was reduced to 350 pounds. If the g was increased to 1.3 g, the load increased to a compressive force of about 1,000 pounds. Projection of these data showed that to reduce the compressive load on the rod to zero, the load factor would have to be reduced to about 0.35 g. Also, the data showed that, as the slat extends, rod loads decrease and change to tensile loads at about 38 percent of rod extension. Beyond 38 percent extension, the rod is subjected to tensile

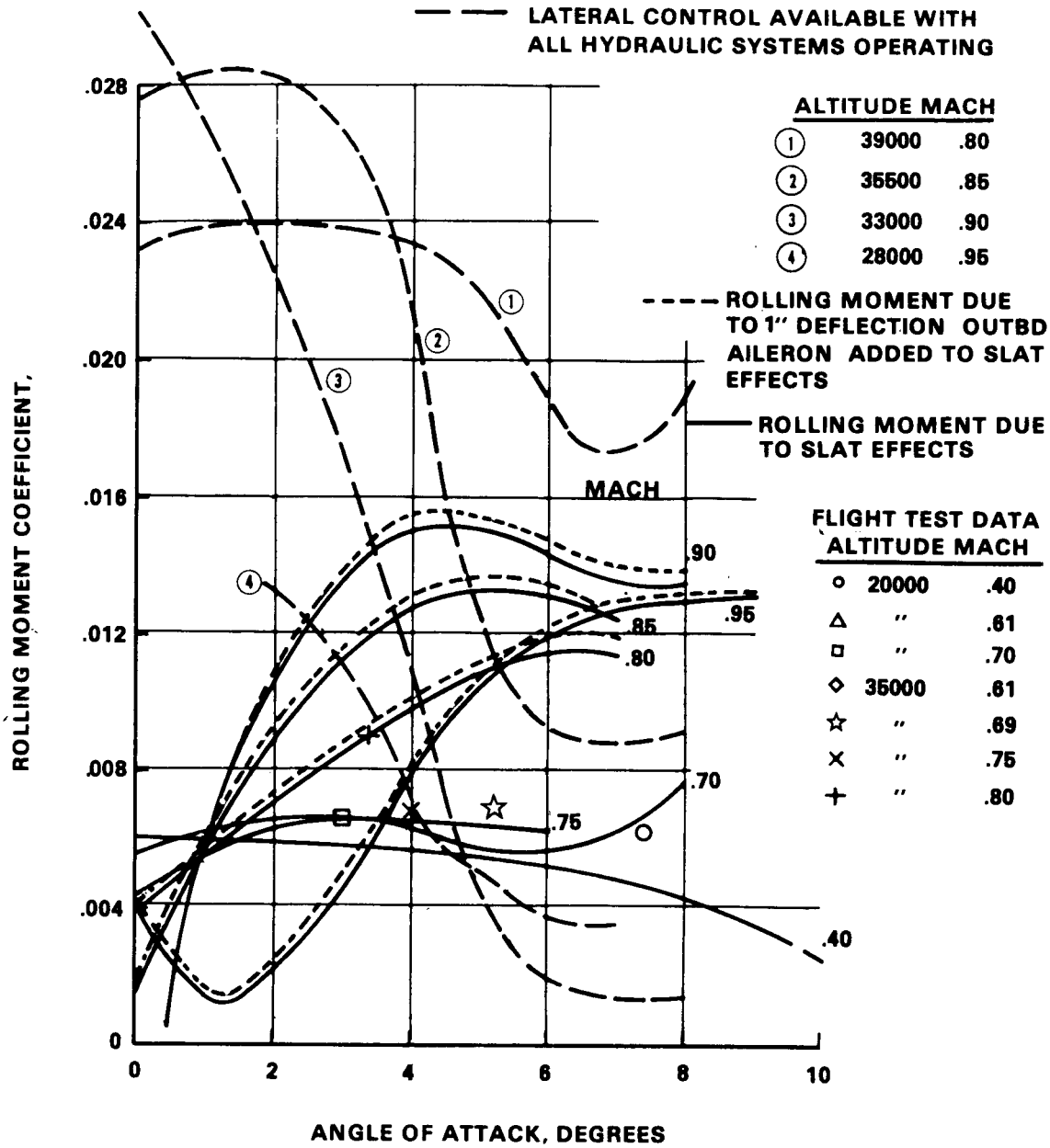


Figure 4.—Rolling moments from extended No. 7 leading edge slat -- adjusted to flight test data points.

loads. At Flight 841's initial conditions, the No. 7 slat actuator rod would have been subjected to a tensile load of about 1,000 pounds with the slat fully extended. At 350 knots equivalent airspeed or higher, the air loads holding the slat extended would exceed 2,400 pounds and the slat could not be retracted because the actuator retraction capacity is 2,400 pounds.

If it is assumed that the right outboard aileron hinge fitting bolt was broken before the No. 7 slat extended, the corresponding aileron float would have produced about a 9 percent reduction in the aerodynamic loads on the No. 7 slat actuator rod. In level cruise flight, this condition would have reduced the compressive load on the rod about 63 pounds.

1.17.5 TWA Flight Operations Safety Bulletin 79-3

On August 6, 1979, TWA issued Flight Operations Safety Bulletin 79-3. The bulletin provided flightcrews with information about B-727 flight characteristics with an extended No. 7 leading edge slat, and it included operational guidance from Boeing OMB 75-7, as revised, concerning an asymmetric slat condition.

1.18 Useful or Effective Investigative Techniques

In an effort to more accurately read FDR foil g-traces when they exhibit high frequency oscillations, a new technique was developed. This technique involved making cellulose impressions of the foil traces and photographing the impressions under high (200 power) magnification. The photographs were then joined and time and amplitude scales were calculated and drawn over the composite traces. This technique permitted the illustration of highly accurate g-trace frequencies and amplitudes on a normal scale. In this manner, it was determined that the high frequency g-trace oscillations associated with airframe buffet on the flight test aircraft's FDR and the accident aircraft's FDR were identical at a frequency of 6 cycles/second and an amplitude of $\pm$ 0.05g.

2. ANALYSIS

2.1 The Flightcrew

The flightcrew was properly certificated and was qualified in accordance with existing regulations; however, the captain had requalified in the B-727 only recently and had returned to line flying after having been off duty for about 3 months with a broken ankle. There was no evidence that medical or physiological problems affected their performance. They had a rest period of about 15 hours from the end of their duty on April 3 to the beginning of duty on April 4. This conformed to the requirements of Federal aviation regulations. Additionally, all members of the flightcrew indicated that fatigue was not a factor in their performance.

2.2 Weather

Weather was not a factor in the accident.

2.3 The Aircraft

The aircraft was properly certificated and was maintained in accordance with existing regulations and procedures. There were no significant uncorrected discrepancies in the aircraft's maintenance log, and following the "C" check in early March 1979, there were no recorded discrepancies concerning the aircraft's flight control, autopilot, flap, hydraulic, or flight instrument systems.

According to the flightcrew, all systems functioned properly before and after departure from JFK and while en route. The flaps and slats retracted properly and on schedule after takeoff, and the flap/slat position indicating system displayed no abormal slat or flap condition.

2.4 Extension of the No. 7 Leading Edge Slat

According to FDR information, about 9 minutes after the aircraft arrived at 39,000 feet, it descended to about 5,000 feet in 63 seconds. The flightcrew could not account for the event that precipitated the aircraft's abnormal descent. The captain, who was flying the aircraft, described the descent as an uncontrollable maneuver involving two rolls to the right, the first of which was preceded by a short period of buzzing and light buffeting of the airframe. According to the captain, everything was normal in the cockpit when the light buffeting began.

Except for the No. 7 slat inboard T-bolt and the right outboard aileron actuator hinge fitting bolt, both of which were broken and showed evidence of preexisting fatigue, the investigation disclosed no significant abnormalities or malfunctions in any of the aircraft's systems. All of the possible sources of uncommanded flight control movements were tested and they functioned properly. The aircraft was operating within its approved performance and maneuvering envelope, and the fuel load was balanced. Also, according to the FDR and the flightcrew, there was no turbulence that might have reduced the aircraft's maneuvering buffet margin, nor were there any significant attitude changes before or during the initial oscillations of the g-trace that would have reduced the aircraft's maneuvering buffet margin.

Since there was no evidence of significant abnormalities in the aircraft's systems or flight controls, operating envelope, or environment and, since the No. 7 leading edge slat was missing from the aircraft, the Safety Board focused its investigation on the possibility that an unscheduled extension of the slat might have caused the uncontrollable maneuver. The manufacturer's flight and wind tunnel test data and the flight simulator tests verified that under Flight 841's cruise conditions at 39,000 feet, a right roll develops with the No. 7 leading edge slat in the extended position and with all other leading edge devices retracted. Additionally, these test data established that under certain combinations of mach number and angle of attack, the rolling moments produced by an extended No. 7 slat would exceed counter moments available from lateral controls and the aircraft would become laterally uncontrollable. Flight simulator tests produced two flight maneuvering situations in which control was lost and in which the simulator traces for airspeed, altitude, g, and heading (after correction for heading gyro gimbal errors) reasonably approximated the accident aircraft's FDR traces for these parameters. Simulations of other flight control system malfunctions produced no reasonable correlations with the FDR traces. Although additional heading gyro tests and gimbal error calculations established that one simulated maneuver (double roll reversal) did not reasonably approximate the accident aircraft's heading trace, the data obtained assisted in establishing lateral controllability margins and was useful in determining the aircraft's probable maneuver.

Based on the physical evidence, aerodynamic data, and the flight simulations, the Safety Board concludes that an extended No. 7 leading edge slat on the aircraft's right wing caused lateral control problems which preceded the aircraft's rapid descent. Additionally, based on the similarities between the accident aircraft's g-trace and the flight test aircraft's g-trace with the Nos. 2, 3, 6, and 7 leading edge slats extended, the Safety Board concludes that the No. 7 leading edge slat on the accident aircraft (and

possibly other slats) began to extend about 2147:32. Also, we conclude that extension of the slat (or slats) created a "buzzing" noise or slight buffet followed by moderate buffet.

Since 1974, the officially recorded history of leading edge slat problems in the B-727 disclosed only one instance of an unscheduled extension of a leading edge slat in flight and one instance of a scheduled extension of multiple slats. All other failures of the slats to extend or retract have occurred during scheduled extensions or retractions. Of the two known instances of slat extensions in flight since 1974, one occurred as a result of an inadvertent but scheduled extension of all the leading edge devices and the other was an unscheduled extension which was caused by a failure of the actuator mount fitting. Neither of these instances resulted in a significant aircraft control problem. Because of the absence of such problems, considerable investigative effort was expended to determine why, out of eight leading edge slats, the No. 7 slat, got isolated in the extended position.

According to the flightcrew, before and immediately after the buzzing began, they saw no lights in the cockpit that indicated an unlocked leading edge device or a failure of a hydraulic system, including the "A" system. Also, the captain stated that there was no inadvertent or deliberate movement of the flap control handle or other controls that would have caused leading edge devices to extend. Therefore, if the flightcrew's recollections are accurate, the No. 7 leading edge slat would have had to extended as a consequence of defects or malfunctions in the No. 7 slat extension/retraction systems.

During the investigation, after repairing and plugging ruptured "A" system hydraulic lines, both the normal and alternate flap control systems were tested. There was no evidence of any malfunction in these systems that might have caused an extension of one or more leading edge devices. Also, the flap/slat indicator system functioned properly--the No. 7 actuating and indicating system could not be checked because the slat was missing, the actuator was broken, and its lines were plugged for the tests.

A fault analysis of the B-727 leading edge slat actuating system indicates that, except for a separation of the piston rod from the actuator piston or a fracture of the piston, at least two failures involving the slat actuator must precede an unscheduled extension of a slat. This is because when the leading edge slat is in the retracted position, the slat actuator piston is held continuously in the locked position by 3,000 psi of "A" system hydraulic pressure and by a mechanical locking device. Therefore, to nullify these features, hydraulic pressure must be lost and the mechanical locking device must fail. There is no evidence that "A" system hydraulic pressure was lost before the slat extended--in fact, the flightcrew statements indicate that it was not lost before the slat extended. Moreover, the landing gear uplocks and the landing gear, which operate off "A" system pressure, retracted and extended respectively, about 72 seconds after the buzzing began. Therefore, any loss or significant reduction of retraction pressure would have had to have involved either a main piston seal failure or a locking piston seal failure. A main piston seal failure would have permitted fluid to flow past the seal to the extend side of the piston, out the return ports, and back into the "A" system reservoir. If the return ports were plugged, the pressure on both sides of the piston could equalize but the locking keys would remain in place from friction and spring pressure. If the locking piston seal also failed, the piston would still maintain its position by friction and spring pressure. Therefore, even a massive leak across either or both piston seals would not have released the mechanical lock.

A locking device failure would have had to involve excessive friction in the switch actuator pin guide while the actuator piston was out of the retract position, thereby freezing the pin in the unlocked position. This condition would have prevented the lock keys from being forced into the lock detent when the actuator piston moved into the retracted position. This condition alone would not have affected slat extension or retraction; however, the amber "in transit" lights on the cockpit displays would have remained illuminated following slat retraction. Therefore, the two concurrent prerequisites for nullification of the hydraulic and mechanical extension restraints would had to have involved a loss of "A" system pressure in conjunction with the freezing of the switch actuator pin in the unlocked position. Both events would have caused warning lights to illuminate in the cockpit.

The forward two-thirds of the actuator cylinder, the actuator piston, the piston rod, the locking pistons, and the locking pins were missing; therefore, the condition of the piston, its seals, and the piston rod could not be verified. However, as described above, it is not likely that even a massive failure of either or both piston seals would have nullified the mechanical lock.

A transverse fracture of the actuator piston would have permitted a maximum differential hydraulic force of about 780 pounds to act on the fractured face of the forward portion of the piston. This force could have extended the No. 7 slat since it was slightly greater than the force produced by air loads on the slat during Flight 841's cruise conditions. However, calculations related to such a fracture in the area of the main piston seal indicate that under normal operational loads, which are predominently axial, a 11,300 percent margin of safety existed at minimum material strength with regard to internal stress levels that could have produced a fracture in that area. Also, under design limit load conditions, a 3,200 percent margin of safety existed. Further, because of locking piston seal frictional forces, 70 or more g's would have been required to unlock the aft portion of such a fractured piston. Calculations for margins of safety regarding a transverse fracture of the main piston in the area of the retract locking keys, establish that since the cross sectional area of the piston in that location is 59 percent of the main seal cross sectional area, operational and design limit safety margins would have been 6,667 and 1,888 percent, respectively. Also, when in the retracted position, there were no forces acting on the piston that would have tended to produce a tranverse fracture in the area of the locking keys. Finally, in 16 years of service history and over 36-million flight hours, such fractures of a slat actuator piston have never occurred. Therefore, the Safety Board concludes that a transverse fracture of the actuator piston was highly improbable.

Even in the remote possibility of one or two failures within the slat actuator that might have nullified both the hydraulic and mechanical restraints, a third condition was necessary to permit an unscheduled extension of the slat. This condition involved aerodynamic loads on the slat that would have produced a tensile force on the actuator piston rod. The evidence indicates that under the aircraft's flight conditions, a compressive force, rather than a tensile force, was acting on the rod when the slat extended.

Based on FDR data and flight test data, the Safety Board concludes that the No. 7 leading edge slat and possibly other slats extended about the time the vertical acceleration trace began to oscillate at an amplitude of about ± 0.05 g. At that time, the aircraft's equivalent airspeed was about 229 knots (245 knots indicated) and its mach number was about 0.80, and according to the manufacturer's flight test and wind tunnel test data, the airloads on the slat would have subjected the slat actuator rod to a compressive load of about 700 pounds, and about 9 percent less if the right outboard aileron was floating. Moreover, the data indicates that the aircraft's vertical acceleration would had to have decreased from 1.0 to about 0.35 g to reduce the

compressive load to zero. The aircraft's vertical acceleration trace does not show a g-reduction of such a magnitude preceding slat extension. Therefore, the third condition needed for unscheduled extension was not met.

The only reasonable single failure within the slat actuator that would have freed the slat to move in opposition to airloads on the slat was a separation of the piston rod from the actuator piston. Separation of the rod from the piston would have permitted the hydraulic pressure within the cylinder to act on the end of the piston rod with 1,545 pounds of force, which could have forced an unscheduled extension of the slat since the aerodynamic loads on the slat produced less than 1,545 pounds of compressive force on the bearing end of the rod. However, the evidence indicates that the piston rod did not separate from the piston, but rather that the bearing end of the rod fractured in overload and remained attached to the slat. The actuator cylinder was fractured and the retained portion of the cylinder bore was distorted to the extent that the actuator piston could not have been in the retracted position when the cylinder broke; otherwise, the piston would have remained in the retained portion of the cylinder. Therefore, the Safety Board concludes that the piston rod was attached to the piston when the slat extended and when the slat was broken from the aircraft.

A metallurgical examination disclosed that the No. 7 slat inboard T-bolt was significantly weakened by fatigue before it failed. Also, the wear pattern on the slat alignment hooks indicate that the slat was misaligned and that the T-bolt may have broken before the slat extended in flight, causing the inboard end of the slat to sag slightly. The precise aerodynamic effects of such a condition could not be determined. However, since compressive airloads existed on the slat in flight and since the slats retracted properly after takeoff, the Safety Board believes that the loads kept the slat essentially aligned with the leading edge of the wing while the slat was in the retracted position. Under such conditions, it is not likely that the fractured T-bolt caused an aerodynamic problem because the slat tracks and piston rod bear almost all of the slat loads while it is retracted. However, once extended, it is probable that the misalignment caused side and friction loads which, in addition to the high tensile load on the rod, exceeded any available hydraulic force for scheduled retraction.

Postaccident investigation disclosed 1 3/32 inch of free play in the right outboard aileron which was attributed to a broken bolt on the aileron actuator's hinge fitting. A metallurgical examination disclosed that the bolt had also failed predominantly in fatigue. However, it could not be determined when the bolt failed. If it failed before the No. 7 slat extended, the resulting free play would have permitted the aileron's trailing edge to float upward about 1 inch which would have produced a localized reduction in the angle of attack and a resulting loss of lift over the wing area forward of the aileron. According to the aircraft manufacturer, such a loss of lift would have caused a rolling moment to the right which would have required about 13° of control wheel displacement to the left to counter. This amount of deflection would have been noticeable if the bolt had broken any length of time before the No. 7 slat extended. However, even assuming that the bolt broke just before or just after slat extension, rather than later in the maneuver, adding the resulting rolling moment of the free play to the rolling moments generated by the slat, sufficient lateral control would have been available initially to restore and maintain wings-level flight.

The evidence involves a fundamental conflict between the flightcrew's statements and the possibilities and probabilities of an unscheduled extension of the No. 7 slat. Although portions of the slat actuator were not found, the evidence indicates that the possibility of a series of malfunctions and failures occurring which permitted the slat to extend aerodynamically or hydraulically is extremely remote. On the other hand, we

recognize that if the No. 7 slat did not extend as the consequence of some series of failures and malfunctions in the slat system, then it must have been extended as a result of flightcrew action.

After carefully weighing all evidence related to this accident, the Safety Board concludes that there is no evidence of any probable combination of failures and malfunctions in the aircraft's flight control system that would have caused an unscheduled extension of the No. 7 leading edge slat. Moreover, since the airspeed decrease which followed extension of the Nos. 2, 3, 6, and 7 slats during flight tests compares almost exactly with the airspeed decrease experienced by Flight 841 following initial oscillation of its g-trace, which under constant thrust and 1.0-g flight conditions can only be attributed to similar drag producing configurations, the Safety Board concludes that the Nos. 2, 3, 6, and 7 slats were extended as a consequence of flightcrew action. Further, that when scheduled to retract by the flightcrew, the No. 7 slat failed to retract probably because tensile forces created by aerodynamic loads combined with friction and side forces on the piston rod, caused by misalignment of the slat, exceeded the available retraction force.

2.5 Loss of Aircraft Control

All of the pertinent evidence indicates that with the No. 7 slat extended, the B-727 is controllable under Flight 841's cruise flight conditions at 39,000 feet and that the aircraft does not become laterally uncontrollable until certain combinations of mach number and angle of attack are exceeded. These combinations are shown in figure 4, and they represent the lowest values, including the adverse effect of a 1-inch upward float of the right outboard aileron. By interpolation, the lowest combination of mach and angle of attack at which the aircraft was laterally uncontrollable was mach 0.83 and an angle of attack of 6°. These are considered the critical controllability values. When mach number or angle of attack was below these values, the aircraft should have been controllable laterally and when exceeded, the aircraft would have been uncontrollable laterally.

The reason for the lack of controllability when the critical values are exceeded is the severe disruption of airflow and associated loss of lift that occurs over the wing area aft of the extended slat. The loss of lift creates a rolling moment that exceeds the countermoment the pilot can produce with full deflection of lateral controls. The aircraft will respond to the imbalance of rolling moments by rolling uncontrollably toward the wing with the extended slat. By omitting any effects of sideslip, the calculated values of mach number and angle of attack established that the accident aircraft was laterally uncontrollable after 2148:07. At that time, the aircraft's indicated airspeed was about 270 knots (0.83 mach), its altitude was about 36,500 feet, and its rate of descent was in excess of 34,000 feet per minute.

After the No. 7 slat was torn from the aircraft, lateral control was restored and the captain was able to roll the wings parallel to the horizontal and recover from the spiral dive. Simulations of the spiral dive confirmed that loss of the slat restored lateral control and made recovery possible. Also, simulations indicated that although extension of the landing gear significantly reduced the aircraft's speed (and mach number), recovery would have been doubtful without loss of the slat because of the high angles of attack which were developed during the latter part of the descent.

Because of the conflict between the captain's assertions regarding the ineffectiveness of lateral controls and the aerodynamic evidence related to aircraft performance and controllability, significant efforts were made to determine the aircraft's actual motions and performance, as reflected in its FDR traces, subsequent to the first

observed anomalies in the g-trace. These efforts included additional heading gyro tests, extensive calculations of gimbal error associated with heading gyro performance, and flight tests.

Analysis of flight test and flight simulator data indicated that high rates of heading change, such as the 5° in 0.5 second that occurred about 2147:45 in the accident aircraft's heading trace, could not be achieved unless the aircraft was in a banked attitude with high load factors applied. Since the load factor was about 1.0 g at that time, the rapid heading change could not have been associated with turning flight. Also, during flight tests, a full rudder deflection sideslip, when released, did not produce such a rapid heading change, and during sudden extensions of the No. 7 slat in flight simulations, the heading changed comparatively slowly to the right. However, the heading gyro tests and heading gyro gimbal error calculations showed that such an apparent rate of heading change could have been produced by a rapid roll to the left. Consequently, it was hypothesized that during the preceding 12-second period between 2147:33 and 2147:45, the aircraft had rolled to the right.

Based on heading gyro tests and actual heading, bank, and pitch angles recorded during flight simulations of Flight 841's maneuver, it was determined that indicated heading could be accurately calculated by using the standard mathematical equation for heading gyro errors. By use of pitch angles and actual heading angles from the simulations, calculations of gimbal error associated with various bank angles produced an indicated heading trace which was comparable to the accident aircraft's FDR trace with the time shift removed. Calculations of bank angles up to 60° were further confirmed by using aircraft turning performance equations and FDR values of normal acceleration, altitude, and airspeed. These data were programed into the Safety Board's scientific data reduction and plotting computer which drew the results for the first 360° of roll shown in figure 5.

As shown in figure 5, the calculated indicated heading trace for these heading, pitch, and bank angles compares almost exactly with the accident aircraft's FDR heading trace with the time shift removed. Beginning about 2147:34, the aircraft began to roll slowly to the right, and about 6 seconds later, the rate of roll began to increase. At 2147:45, the aircraft was in a right bank of about 35°, after which it was rolled rapidly to the left to near a wings-level attitude. About 2147:47, the aircraft began to roll again to the right, and the roll was arrested briefly at 2147:51 near 35° of right bank. At 2147:53, the aircraft resumed its roll to the right and about 2148:07, the aircraft was inverted with a pitch attitude about 45° below the horizontal. As shown in figure 5, the aircraft's actual heading (calculated) changed comparatively little during the 360° of roll.

Based on flight test data and known times for extension and retraction of leading edge slats, the 12-second period between 2147:33 and 2147:45 was examined to determine whether a compatible relationship existed among slat extension/retraction cycles, slat asymmetry, and calculated angles of bank that were achieved during the period. According to flight tests, buffet begins and is shown in the g-trace about 2 seconds after the beginning of leading edge slat extension. Therefore, it appears that slat extension began about 2147:32. Assuming a reaction time of 2 seconds to retract the slats, retraction would have begun about 2147:39 and would have been completed 5 to 6 seconds later. This would mean that slat asymmetry would have begun about 2147:40 which would have caused an increasing rate of roll to the right while the Nos. 2, 3, and 6 slats retracted. Gimbal error calculations indicate that from 2147:40 to 2147:45 the aircraft rolled from about 10° of right bank to about 35° of right bank at an increasing roll rate. This is compatible with the increasing slat asymmetry that would have occurred during that 5 seconds.

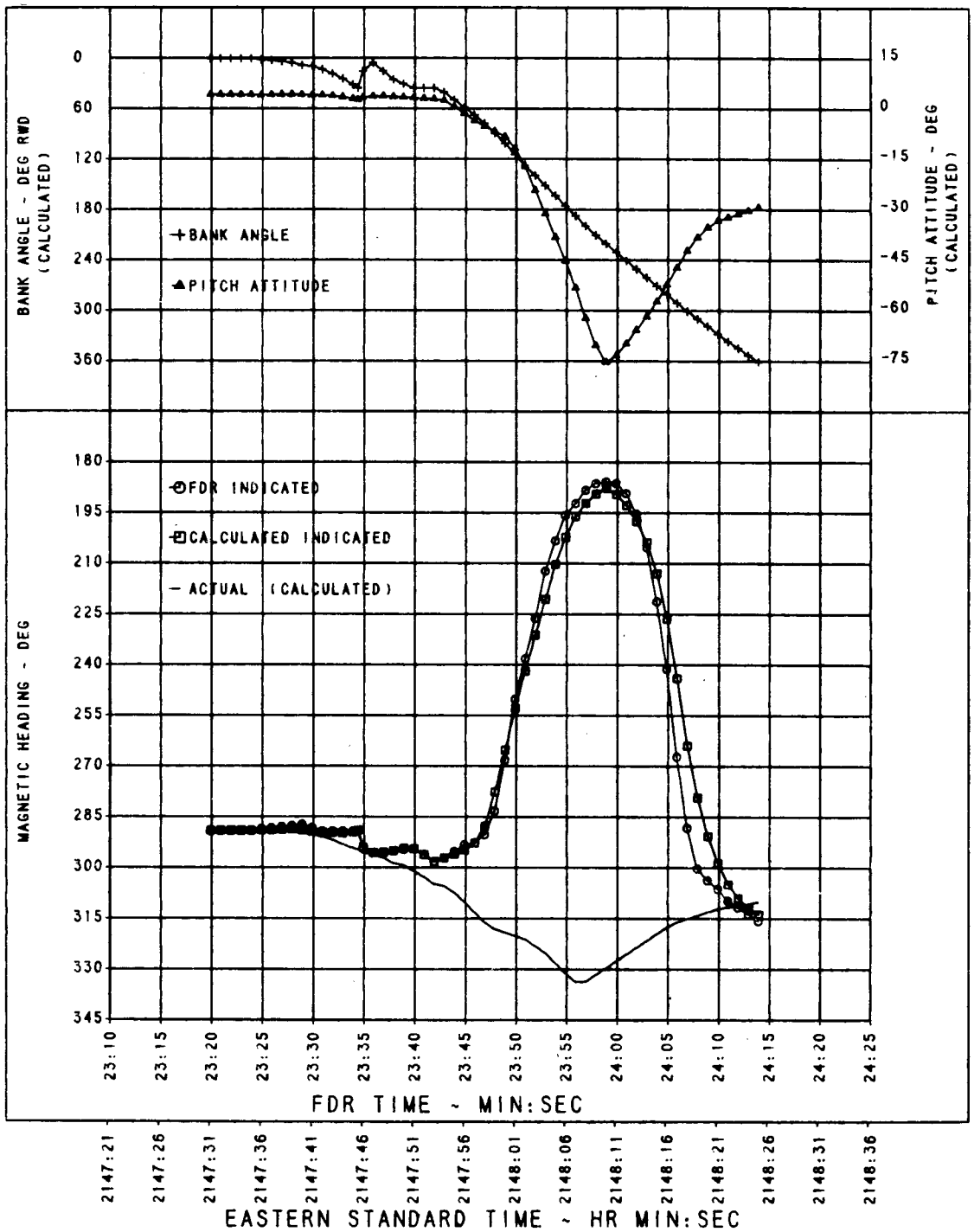


Figure 5.—Comparison of calculated and FDR headings.

Given the foregoing assessment of the aircraft's motions and performance, it is apparent that the aircraft was initially controllable following isolation of the No. 7 leading edge slat in the extended position because, beginning about 2147:45, the aircraft was rolled to the left to a near wings-level position. About 4 seconds later, the aircraft was again banked to the right about 35°. However, following a 2-second pause at 35°, the aircraft resumed its roll to the right and it began to descend rapidly.

The Safety Board is not able to determine conclusively why the captain failed to retain control of the aircraft after having once rolled it from a 35° right bank to near a wings-level attitude. Although we cannot positively exclude spatial disorientation of the captain as a possible reason for his failure to retain control, we believe it more probable that a number of factors combined to place the aircraft in an attitude where critical controllability parameters were exceeded well before the parameters established from subsequent flight simulations of the maneuver. These factors involve the actual effectiveness of the lateral controls, the actual margins of lateral control with an extended No. 7 slat, a cruise mach number that we believe was higher than 0.80, the effects of sideslip induced by full deflection of lateral controls and rudder, and distraction of the captain.

It is possible, for instance, that because of aileron and spoiler rigging tolerances, the accident aircraft had less than specified left roll capability. Although these tolerances would not have been noticeable in normal maneuvering flight, they could have become a factor during full deflection of the controls. Also, it is possible that lateral control margins were reduced to values similar to those shown in figure 4, rather than the higher values used in the flight simulations.

The Safety Board believes that Flight 841's cruise mach number at 39,000 feet pressure altitude probably was higher than 0.80. Support for a higher cruise mach number is indicated by a comparison of the airspeed and altitude traces with the captain's statements about mach number and airspeed. He said that he climbed the aircraft from 35,000 to 39,000 feet at 0.80 mach and leveled at 39,000 feet at 0.80 mach. According to the FDR, when the aircraft was leveled at 39,000 feet its indicated airspeed was 240 knots. However, at that pressure altitude, indicated airspeed must be 247.5 knots to achieve 0.80 mach. Consequently, it appears that the recorded airspeed was 7.5 knots too low but within recorder tolerances of $\pm$10 knots. The captain also stated that just before the buzzing began the indicated airspeed was 252 knots. At that time, the recorded airspeed was about 245 knots which tends to confirm a recorder error of -7.5 knots. At an indicated airspeed of 252 knots at 39,000 feet, the aircraft would had to have been at mach 0.816, rather than 0.80. Additionally, when the -7.5 error is accounted for during Flight 841's maneuver, higher mach numbers were achieved than those originally calculated and, at certain times, the aircraft would have been closer to its critical controllability mach number than was originally calculated.

Under conditions of slat asymmetry and high mach numbers, the effects of sideslip on lateral stability and control can be significant even though the aircraft is well below its freestream critical controllability mach number. For instance, at a freestream mach number of 0.83, 6° of angle of attack, and 0° of sideslip, the local critical controllability mach number at the leading edge of the No. 7 slat (extended) is 0.688 because of the 34° sweep angle of the wing. If a 5° right sideslip is introduced, the local critical mach number is reached at a freestream mach number of 0.787. Consequently, with the introduction of sideslip, the accident aircraft could have reached critical controllability parameters at freestream mach numbers significantly below the critical values for 0° of sideslip.

Flight test data and flight simulations indicated that in the B-727 full deflection of lateral controls and full deflection of rudder in the same direction can produce sideslip angles of 4.5° to 6.5°. Also, flight tests in conditions similar to Flight 841's at 39,000 feet showed that rolls with full deflection of the lateral controls produced sideslip angles of about 5° in the direction opposite to the roll. As shown in figure 5, about 2147:47, after a left roll to a near wings-level position, the aircraft again began to roll to the right and about 4 seconds later was at 35° of right bank where the roll was checked for a few seconds. Under the circumstances, we believe this roll probably occurred while the captain was distracted by activities related to the No. 7 slat having been isolated in the extended or partially extended position. Thereafter, if full lateral and rudder controls were applied simultaneously or in rapid succession to stop the roll, significant sideslip could have been introduced at a critical time, and the aircraft could have become laterally uncontrollable well before its 0° sideslip controllability parameters were reached.

Calculations of mach number and angle of attack, which take into account a recorder error of -7.5 knots and the effects of roll rate on angle of attack, indicate that at 2147:51, the accident aircraft was at mach 0.79 and an angle of attack of 5.7°. Also, according to gimbal error calculations, the aircraft was banked about 35° to the right. Under these conditions, if a 4.8° right sideslip angle was introduced, lateral control could have been lost. About 2147:54, the aircraft's rate of descent began to increase very rapidly, although FDR indicated airspeed was stable at 236 knots, which indicates that thrust was substantially reduced or drag was substantially increased shortly before that time. According to the captain, he reduced the throttles to flight idle well before he extended the speed brakes. Consequently, the Safety Board believes that shortly before 2147:54, the captain removed his right hand from the control wheel and used his right hand to retard the throttles to flight idle. Moreover, we believe that the aircraft was then in a substantial sideslip condition which, perhaps in conjunction with some relaxation of the lateral controls or less than optimum left roll authority, caused the aircraft to exceed its critical controllability parameters and to roll uncontrollably into a rapid spiralling descent.

As stated before, we are not able to fully explain why the loss of control occurred. However, we note that the foregoing explanation is consistent to some degree with the captain's statements about his manipulation of flight and throttle controls. Also, we believe that under the circumstances, after having apparently controlled the initial roll to the right, it would not have been unusual for the captain to have diverted his attention from the flight instruments to other instruments and controls in an effort to determine the cause of the initial roll and the cause of the continuing airframe buffet, particularly since the other crewmembers apparently were not aware initially of the aircraft's condition.

As shown in figure 5, the second roll to a 35° right bank occurred in about 4 seconds—a comparatively brief period in which even a slight distraction could have been critical. At the conclusion of the 4 seconds, the roll was stopped for a few seconds which indicates that lateral controls probably were applied quickly and fully in response to the comparatively rapid rate of roll. Since the captain followed the application of lateral controls with a significant amount of rudder, as indicated by his statements, we conclude that a sideslip condition was generated which placed the aircraft in a laterally uncontrollable condition as evidenced by the resumption of the roll to the right. Further, it is possible that cocking or deflection of the No. 7 slat added to rolling moment imbalance at this critical point. Thereafter, the speeds and angles of attack generated by the rapid descent and high g-forces combined with the extended No. 7 slat to keep the aircraft in an uncontrollable condition until the slat was torn from the wing.

During the investigation, questions were raised about why the flightcrew might have extended the leading edge slats under the existing operating conditions. Several theories were considered, including accidental actuation of the flap lever, maloperation of the alternate flap system, and an unsuccessful attempt to extend trailing edge flaps independently of leading edge slats, possibly in an effort to improve aircraft performance.

The flightcrew denied having moved any controls that would have caused extension of flaps or slats. Since there is no other available evidence of flightcrew activities in the cockpit, the Safety Board is not able to determine conclusively why the Nos. 2, 3, 6, and 7 leading edge would have been extended. However, we note that since the flap lever must be moved up and over a gate before it can be moved to a flap/slat extension position, it is not likely that the lever was moved accidentally. Further, since operation of the alternate flap system to extend leading edge devices results in random and initially unsymmetrical extension of leading edge flaps and slats, extension of only the Nos. 2, 3, 6, and 7 slats would not have been likely.

In summary, the Safety Board concludes that the following sequence of actions and events probably occurred to cause Flight 841 to enter an uncontrollable spiral dive involving two 360° rolls and a loss of about 34,000 feet of altitude in about 63 seconds:

- While cruising at mach 0.816 and 39,000 feet pressure altitude and with the autopilot controlling the aircraft, an attempt was made to extend 2° of trailing edge flaps independently of the leading edge slats, probably in an effort to improve aircraft performance.

- The attempt to independently extend 2° of trailing edge flaps was not successful, and about 2147:32 the Nos. 2, 3, 6, and 7 leading edge slats began to extend. Two seconds later, the aircraft began to buffet and roll slowly to the right. Six to seven seconds later, the rate of roll began to increase due to increasing slat asymmetry as the Nos. 2, 3, and 6 slats retracted. The No. 7 slat failed to retract.

- About 2147:45, the aircraft reached about 35° of right bank where the captain disconnected the autopilot and rapidly rolled the aircraft to the left to a near wings-level attitude. The aircraft could have been stabilized in wings-level flight with appropriate deflection of the lateral controls.

- About 2147:47, the aircraft again began to roll to the right, probably while the captain was distracted by activities related to the isolation of the No. 7 slat in the extended position.

- Shortly before 2147:51, the captain recognized the rapid right roll, and he rapidly applied full deflection of the lateral controls to stop the roll. The roll was stopped near 35° of right bank for several seconds during which the captain removed his right hand from the control wheel, pulled the throttles to flight idle, and deflected full or nearly full left rudder to augment lateral controls.

- In response to the rapid and full or nearly full deflection of the flight controls, the aircraft entered a substantial right sideslip. The sideslip combined with the aircraft's mach number and angle of attack to reduce the lateral control margin to zero or less. The

aircraft resumed the right roll and began to descend rapidly and uncontrollably. The captain extended speed brakes, detected no reaction, and retracted them.

- About 2148:25, the aircraft completed 360° of roll while descending to about 21,000 feet. Shortly thereafter, the captain commanded landing gear extension which was accomplished by the first officer. The aircraft continued to descend rapidly, and it continued to roll to the right until the No. 7 slat was torn from the wing and lateral control was restored. About 2148:58, the captain regained control of the aircraft at an altitude of about 8,000 feet.

Since our weighing of the evidence involves a rejection of the possibility of an unscheduled extension of the No. 7 slat and a partial rejection of the captain's recollection of his actions following extension of the slats, the Safety Board believes that the following comments are appropriate: We believe the captain's erasure of the CVR is a factor we cannot ignore and cannot sanction. Although we recognize that habits can cause actions not desired or intended by the actor, we have difficulty accepting the fact that the captain's putative habit of routinely erasing the CVR after each flight was not restrainable after a flight in which disaster was only narrowly averted. Our skepticism persists even though the CVR would not have contained any contemporaneous information about the events that immediately preceded the loss of control because we believe it probable that the 25 minutes or more of recording which preceded the landing at Detroit could have provided clues about causal factors and might have served to refresh the flightcrew's memories about the whole matter.

3. CONCLUSIONS

3.1 Findings

1. The flightcrew was properly certificated and was qualified for the flight; the captain had requalified in the B-727 only recently, had flown 21 hours 50 minutes since requalifying, had not flown for about 3 months before requalification, and had flown exclusively as a first officer on B-747's from November 1977 to December 1978.

2. The aircraft was certificated and maintained in accordance with existing regulations and procedures.

3. The inboard slat track T-bolt on the No. 7 leading edge slat had failed predominantly in fatigue.

4. The right outboard aileron actuator hinge fitting bolt rib had failed predominantly in fatigue.

5. The wear pattern on the slat alignment hooks indicated that the No. 7 leading edge slat was not aligned properly.

6. There was no other evidence of irregularity, malfunction, or failure of the aircraft's flight control, autopilot, hydraulic, or flap systems that might have caused or contributed to a lateral control problem.

7. The aircraft's gross weight and center of gravity were within the authorized performance and maneuvering envelopes when a lateral control problem developed.

8. The aircraft was cruising at 0.816 mach in level flight and smooth air at 39,000 feet when a lateral control problem developed.

9. A failure of the right outboard aileron actuator hinge fitting bolt before development of the lateral control problem would have permitted the aileron to float upward about 1 inch; this condition would have required about 13° of left deflection of the control wheel to maintain wings-level flight and would have been noticeable.

10. A right roll and a lateral control problem were caused by isolation of the No. 7 leading edge slat in the extended position.

11. There was no evidence of any combination of failures or malfunctions in the aircraft's flight control system that would have caused an unscheduled extension of the No. 7 leading edge slat.

12. The Nos. 2, 3, 6, and 7 leading edge slats were scheduled to the extended position, and the Nos. 2, 3, and 6 slats were retracted as a consequence of the flightcrew's actions.

13. When scheduled to retract, the No. 7 leading edge slat failed to retract probably because tensile forces created by air loads combined with friction and side forces on the piston rod, caused by preexisting misalignment of the slat, exceeded the available hydraulic retraction force.

14. The No. 7 leading edge slat in the extended position created rolling moments to the right that could have been countered with about 46° of control wheel deflection to the left; an additional 13° of control wheel deflection would have been needed to counter moments associated with a 1-inch upward float of the right outboard aileron.

15. After recognizing the right roll condition, the captain rolled the aircraft to a near wings-level upright position; thereafter, through untimely use of the flight controls, he permitted the aircraft to roll to the right into an uncontrollable attitude. The captain probably was distracted immediately after restoring the aircraft to near level flight by his efforts in attempting to rectify the source of the control problem.

16. The captain probably induced sideslip shortly before 2147:54 when the aircraft was at mach 0.79, an angle of attack of 5.7°, and an angle of bank of about 35° to the right. A sideslip angle of 4.8° to the right could have caused the aircraft to become laterally uncontrollable.

17. The aircraft descended in a spiral dive from 39,000 to about 5,000 feet in 63 seconds; during the descent, the aircraft's speed increased to a maximum speed of about 0.96 mach at 31,800 feet.

18. When the aircraft's speed exceeded 0.83 mach and its angle of attack exceeded 6° near 36,500 feet, the rolling moments caused by the extended No. 7 slat substantially exceeded the maximum available lateral control authority at 0° of sideslip.

19. The aircraft was not controllable during its descent below about 36,500 feet until the No. 7 leading edge slat separated from the right wing.

20. Vertical acceleration forces increased throughout the spiral descent to a maximum of about 6.0 g's during the recovery.

21. The accident was survivable.

22. Minor injuries to passengers were caused by the g-forces.

3.2 Probable Cause

The Safety Board determines that the probable cause of this accident was the isolation of the No. 7 leading edge slat in the fully or partially extended position after an extension of the Nos. 2, 3, 6, and 7 leading edge slats and the subsequent retraction of the Nos. 2, 3, and 6 slats, and the captain's untimely flight control inputs to counter the roll resulting from the slat asymmetry. Contributing to the cause was a preexisting misalignment of the No. 7 slat which, when combined with the cruise condition airloads, precluded retraction of that slat. After eliminating all probable individual or combined mechanical failures or malfunctions which could lead to slat extension, the Safety Board determined that the extension of the slats was the result of the flightcrew's manipulation of the flap/slat controls. Contributing to the captain's untimely use of the flight controls was distraction due probably to his efforts to rectify the source of the control problem.

4. RECOMMENDATION

On January 21, 1980, the National Transportation Safety Board issued the following recommendation to the Federal Aviation Administration:

> Disseminate to all B-727 operators and flightcrews information of the type included in Boeing Operations Manual Bulletin 75-7 and Trans World Airlines Flight Operations Safety Bulletin 79-3 which address control problems associated with high-speed asymmetrical leading edge slat configuration on B-727 aircraft. (Class II, Priority Action) (A-80-8)

In a letter dated April 18, 1980, the FAA declined to take any action on the recommendation. In a letter dated June 20, 1980, the Safety Board disagreed with the FAA's position, and the FAA responded by letter dated December 18, 1980, that no action would be taken pending an evaluation of the flight test data acquired in October 1980 and the Safety Board's final report of the accident.

One Board Member filed the following concurring and dissenting statement:

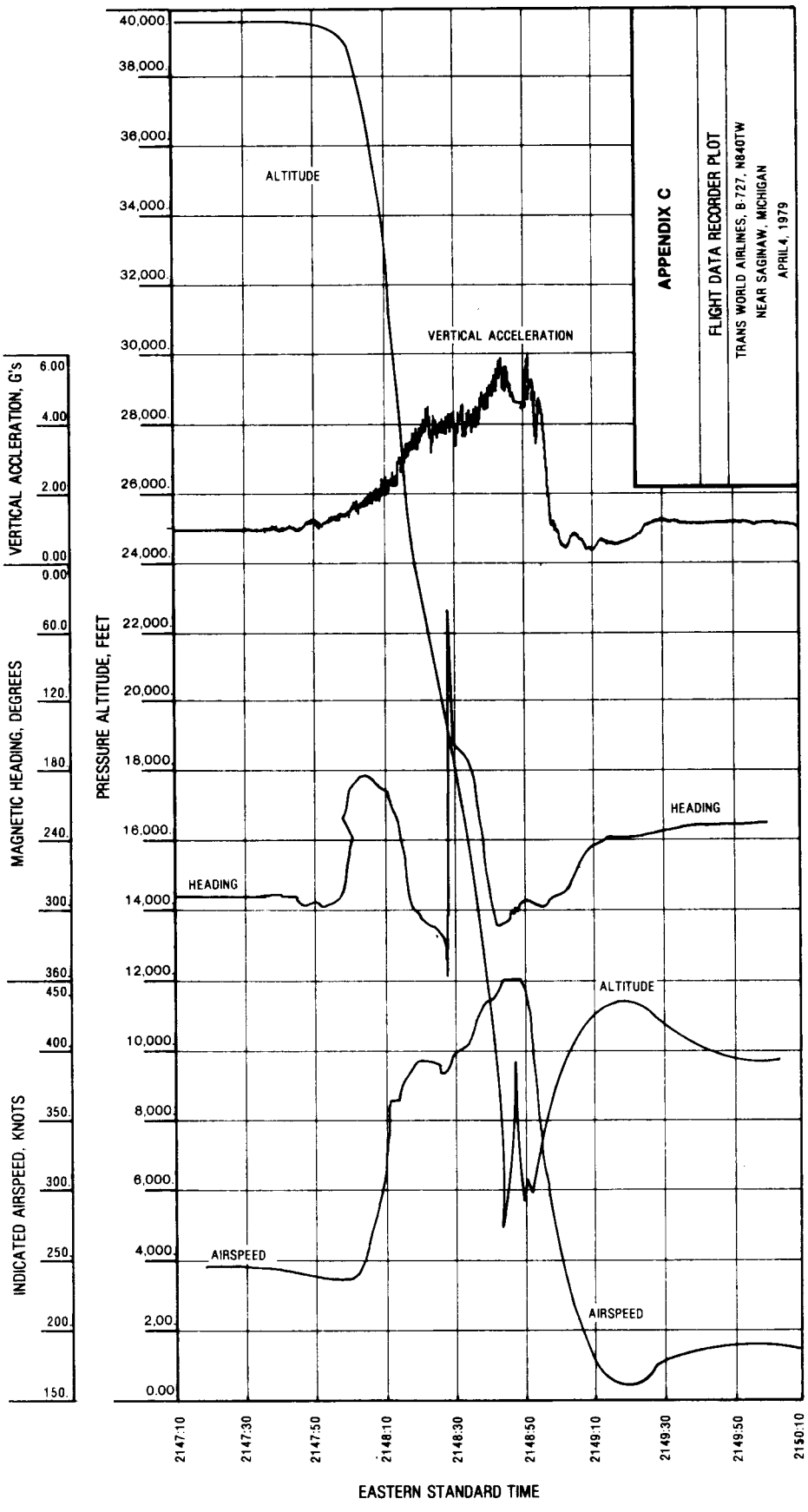
Although I voted to approve the Board's report which concluded that the extension of the leading edge slat was due to flightcrew action, I do so reluctantly.

The report as written, based on the available evidence, i.e., the analysis of the flight data recorder, the simulator tests, the flight tests, and the tilt table tests, appears to support the Board's conclusion. However, I am troubled by the fact that the Board has categorically rejected the crew's sworn testimony without the crew having had the opportunity to be confronted with all of the evidence upon which the Board was basing its findings. At the time of the first deposition, the following evidence was not available to the crew or to the Board: the flight data recorder analysis, the results of the simulator and flight tests, and the tilt table tests. Although the crew was deposed a second time, their testimony was limited to one issue, i.e., the physical location of the flight engineer at the time of the incident. I had recommended that since the Board was ordering a second deposition it be conducted de novo so that the crew would have been aware of all the evidence. The Board did not agree.

Furthermore, I do not agree that a probable cause of this accident, as stated by the Board, was "the captain's untimely flight control inputs to counter the roll resulting from the slat asymmetry." In my opinion, the captain acted expeditiously and reasonably in attempting to correct for the severe right roll condition induced by the extended slat.

ICAO Note: Figs. 2 and 3 and Appendices A, B, D, E and F to the Report are not reproduced.

ICAO Ref: 079/79


APPENDIX C
FLIGHT DATA RECORDER PLOT
TRANS WORLD AIRLINES, B-727, N840TW
NEAR SAGINAW, MICHIGAN
APRIL 4, 1979
PRESSURE ALTITUDE, FEET
40,000.
38,000.
36,000.
34,000.
32,000.
30,000.
28,000.
26,000.
24,000.
22,000.
20,000.
18,000.
16,000.
14,000.
12,000.
10,000.
8,000.
6,000.
4,000.
2,00.
0.00
ALTITUDE
VERTICAL ACCELERATION
HEADING
HEADING
ALTITUDE
AIRSPEED
AIRSPEED
VERTICAL ACCLERATION, G's
6.00
4.00
2.00
0.00
MAGNETIC HEADING, DEGREES
0.00
60.0
120.
180.
240.
300
360.
INDICATED AIRSPEED, KNOTS
450.
400.
350.
300.
250.
200.
150.
2147:10
2147:30
2147:50
2148:10
2148:30
2148:50
2149:10
2149:30
2149:50
2150:10
EASTERN STANDARD TIME

No. 5

Sikorsky S61-L, N618PA, accident at Newark International Airport, New Jersey, United States, on 18 April 1979.
Report No. NTSB-AAR-79-14 dated 27 September 1979, released by the National Transportation Safety Board, United States.

SYNOPSIS

About 1823 e.s.t., on April 18, 1979, New York Airways, Inc., Flight 972 crashed on Newark International Airport while attempting an emergency landing. The flight had just taken off from the airport and was at an altitude of 1,200 ft about 1 mile to the east when one of five blades broke and separated from the tail rotor. As a result, severe vibrations in the tail rotor assembly caused the tail rotor gearbox and rotor assembly to separate from the aircraft when it had descended to about 150 ft above the ground. Without a tail rotor to maintain its stability, the helicopter entered a rapid nosedown, right turn to the ground. Of the 18 persons on board, 3 passengers were killed; 10 passengers and 3 crewmembers were injured seriously.

Metallurgical examination revealed that the tail rotor blade failed after a fatigue crack propagated across 90 percent of the blade's leading edge spar and about 2 in. of the blade skin, which weakened the blade structure.

The National Transportation Safety Board determines that the probable cause of this accident was the separation of the tail rotor assembly and gearbox from the aircraft at an altitude which made further controlled flight impossible. The rotor assembly and gearbox separated because of severe vibrations in the rotor assembly which were induced by the loss of a tail rotor blade due to fatigue failure.

Contributing to the severity of the passengers' injuries were the seat failures which occurred when the deceleration forces exceeded the relatively low design strength of the FAA-approved seats, and the lack of guidance on a passenger brace position for emergency landings.

1. FACTUAL INFORMATION

1.1 History of Flight

On April 18, 1979, a New York Airways Sikorsky S61-L helicopter (N618PA) was being operated as Flight 972 from Newark International Airport, New Jersey, to La Guardia Airport, New York. Flight 972 departed gate 21 at Newark International Airport at 1820 1/ with 3 crewmembers and 15 passengers on board. The first officer was flying the aircraft from the left cockpit seat; both flightcrew members were qualified as captain. At 1821:37, the flight was cleared for takeoff, to climb to 1,400 ft, and to depart eastward to La Guardia Airport. Visual meteorological conditions prevailed at the airport, and the wind was 010° at 15 kns.

At 1822:56, Flight 972 climbed to 1,200 ft and was about 1 mile east of the airport when there was a loud bang, similar to an explosion, followed by severe vibrations in the flight controls. The cockpit voice recorder (CVR) revealed that at 1823:04 the first officer transmitted, "Mayday, mayday, mayday, New York Five is landing on the runway." At 1823:10, the captain transmitted, "I don't know what we've got but we're going to make an emergency landing at Newark." These transmissions were not recorded on the Newark air traffic control tower tape recording, since the first transmission was made on the company frequency and the second was cut out by another transmission. At the same time, the flight attendant cautioned the passengers to insure that their seatbelts were fastened tightly. She later instructed the passengers to remove their eyeglasses.

At 1823:16, the local controller transmitted, "New York Five (Flight 972), you got a problem?" When the first officer responded that they had a control problem, the local controller stated, "Okay you set it down anywhere you want as long as it's not on twenty nine." At 1823:23, the first officer stated, "Yes we're going straight ahead, sir." The captain had instructed the first officer not to make any turns.

The flightcrew stated that as soon as they heard the explosion and felt the vibrations they knew there was a problem with the tail rotor. At first, they thought the drive shaft had failed. The first officer stated that his first instinct was to enter autorotation 2/ and land immediately. However, since that maneuver would have increased rotor rpm and rate of descent, he elected not to autorotate. Additionally, since the maneuver would have intensified the severe vibration, he believed he would not be able to control the aircraft. Since the helicopter was controllable, the crew elected to reduce rpm, slow the airspeed, and attempt to land on the airport.

1/ All times herein are eastern standard time based on the 24-hour clock.
2/ A term for a flight condition during which no engine power is supplied and the main rotor is driven only by the action of the relative wind. It is a means of landing safely after engine failure or certain other emergencies.

According to the first officer, as Flight 972 approached the airport on a heading of about 360°, the airspeed was about 60 kns and the rotor rpm was below 100 percent. The first officer stated that he planned to land beyond the threshold of runway 22R, or near runway 29. Witnesses who first saw Flight 972 near the east boundary of the airport at an altitude of 500 ft stated that the helicopter was "swaying from side to side," and "the tail moved erratically from side to side." (See Appendix D.) As the helicopter passed through about 150 ft, an object—described by witnesses as "a suitcase, or a body or a mailbag"--fell from the left side of the aircraft, and the helicopter's flightpath and attitude changed abruptly. Simultaneously, the helicopter began to turn, according to witnesses, from 90° to 270°; one witness described a turn of two revolutions. The witnesses reported both right and left turns. According to the first officer, at 150 ft the vibrations ceased.

As the helicopter began to turn, its nose went down 35° to 60° and it descended. At that point, the first officer lost all directional control and the aircraft would not respond to rudder pressure. It struck the ground in a nosedown attitude on the right side of the fuselage, it bounced once and fell on its left side. There was no fire or explosion. The helicopter crashed on the grassy area adjacent to runway 22L and came to rest on the west edge of runway 22L, about 610 ft from the threshold.

At 1823:39, the CVR had recorded two loud noises. At 1823:41, the first officer exclaimed, "All right," and at 1823:43 the captain stated, "No tail rotor." As the aircraft descended in a right turn, the captain closed the fuel levers for both engines. Just before the helicopter hit the ground, the first officer attempted to flare it by applying back pressure to the cyclic and pulling up on the collective. The latter maneuver did raise the nose slightly. The first officer stated that the indicated airspeed was about 60 kns when the captain declared, "No tail rotor." Also, before that event, the helicopter's rate of descent never exceeded 600 to 800 fpm. He could not estimate the helicopter's rate of descent near impact, but he stated that its forward speed over the ground was low. The CVR recorded the loss of electrical power at 1823:47.

The accident occurred during the hours of daylight at position 40° 42'N latitude and 74° 10'W longitude. The elevation of the accident site was about 10 ft mean sea level.

1.2 Injuries to Persons

Injuries	Crew	Passengers	Others
Fatal	0	3	0
Serious	3	10	0
Minor/None	0	2	0

1.3 Damage to Aircraft

The aircraft was destroyed.

1.4 Other Damage

Not applicable.

1.5 Personnel Information

The crewmembers were properly certificated and qualified for the flight. (See Appendix B.)

1.6 Aircraft Information

The aircraft was certificated and maintained in accordance with Federal Aviation Administration (FAA) requirements. (See Appendix C.) There were no recent maintenance entries in the aircraft logbook related to the tail rotor assembly, tail rotor gearbox, or any tail rotor blades. The crew reported no unusual vibrations during the three previous flights on April 18.

Flight 972's gross takeoff weight was 16,989 lbs, 1,000 lbs of which was jet A fuel. The maximum allowable gross takeoff weight was 19,000 lbs. Flight 972's center of gravity was 265.3 in. which was within limits.

1.7 Meteorological Information

Surface weather observations taken at Newark International Airport by National Weather Service personnel before and after the accident were:

1751 5,500 ft scattered; visibility — 30 mi.; temperature -- 58°F; dewpoint -- 20°F; wind — 340° at 17 kns; altimeter setting -- 30.04 inHg.

1826 7,500 ft scattered; visibility — 30 mi.; temperature — 56°F; dewpoint-- 21°F; wind -- 360° at 16 kns; altimeter setting — 30.06 inHg.

1.8 Aids to Navigation

Not applicable.

1.9 Communications

No communications difficulties were reported.

1.10 Aerodrome Information

Newark International Airport, elevation 18 ft mean sea level, has four hard surfaced runways. The New Jersey Turnpike parallels the airport on the east side. Flight 972 declared the emergency at Port Newark, an area east

of the turnpike which contains docks, a container loading facility, container storage and semitrailer parking areas, and several warehouses.

1.11 Flight Recorders

Flight 972 was equipped with a Fairchild cockpit voice recorder. The recorder was not damaged. The tape was transcribed, and the quality of the recording was good.

There was no flight data recorder nor was the recorder required.

1.12 Wreckage and Impact Information

Although Flight 972 approached the airport on a northerly heading, the helicopter crashed on a heading of 242°. A 23-ft-long by 8 in.- deep gouge marked the initial impact point. There was little indication of forward movement. Flight 972 struck the ground in a nosedown upright attitude; the nose section, the right landing gear, and right side of the fuselage hit first. The helicopter rolled over and came to rest on its left side. The impact site was about 35 ft west and 1,000 ft beyond the threshold of runway 22L.

The tail rotor gearbox and the attached tail rotor assembly were found about 410 ft west of the main wreckage; an 8-in. section of the tail rotor control rod was found 650 ft west of the main wreckage; two blade skin sections were found about 500 ft west of the main wreckage; and a 35-in. outboard section of one of the tail rotor blades was found 5,100 ft south of the main wreckage site. All other major components were found at the main wreckage site.

The bottom of the fuselage from the nose aft to fuselage station (FS) 391 was crushed upward about 9 in. to waterline (WL) 70. Both main landing gears had separated aft and upward. The tail cone fuselage structure from FS 493 aft was relatively undamaged. The engine cowlings, engines, transmission fairings, and transmissions were not damaged.

Except for one main rotor blade, which separated about 40 in. from its attach point, the remaining four main rotor blades were attached to the rotor hub. The blades showed no evidence of either preimpact distress, damage, or failure. The main rotor head assembly was in good condition, and there was no evidence of any preimpact damage or failure. There was no evidence of any preimpact damage or failure to the control system between the cockpit and the main rotor stationary swashplate and from the rotating swashplate to the rotor blades.

The fuel, hydraulic, and electrical systems showed no evidence of preimpact damage.

The power train, from the engine high speed shafts into the main transmission, and the main transmission were intact. There was no evidence of any internal failure in the transmission, and the main rotor shaft was undamaged.

The tail rotor drive shaft was intact from the main transmission through the intermediate gearbox — which was not damaged — to the laminated coupling located at the input to the tail rotor gearbox. This input coupling had separated in torsion when the tail rotor gearbox separated.

The tail rotor gearbox separated from its mount on the tail pylon, and three of the four attachment lugs separated from the gearbox. The left aft attachment lug separated from the tail pylon but remained attached to the gearbox. Except for the lug separations, the housing was in good condition. The left aft attachment bolt was found 30 ft behind the tail boom at the impact site. When examined, the gearbox rotated freely and the chip detector plug was clean.

The tail rotor assembly consisted of five rotor blades, labeled red, black, blue, yellow, and white and attached to the tail rotor hub. The outboard 35 in. of the black blade was found about 5,100 ft south of the crash site. The remaining four tail rotor blades were attached to the tail rotor assembly. The tail rotor blades which remained attached to the hub had been bent, scraped, and broken during impact; there was no indication of preimpact failure. The black blade fracture surface appeared to be a progressive-type fatigue failure through the blade spar.

1.13 Medical and Pathological Information

A review of the flightcrews medical records revealed no evidence of medical problems that might have affected their performance.

The three crewmembers received serious compression fractures of the lower back, multiple abrasions, contusions, and lacerations. In addition, the captain received a moderate cerebral concussion.

Although seatbelts and shoulder harnesses were installed on the cockpit crew seats, neither pilot was wearing his shoulder harness. The flight attendant's seatbelt was fastened securely.

The three passengers who were killed were seated in seat rows 3 and 4, which were in the area above the main landing gear. Two of these passengers received cervical fractures while the third suffered a hemothorax. Injuries to the other passengers included fractures of the lower extremities, a fractured skull, fractured ribs, multiple abrasions, contusions, and lacerations.

1.14 Fire

Although fuel spilled and formed puddles around the wreckage, there was no fire. The first officer attributed the absence of fire after impact to the fact that the battery switch and fuel had been shut off just before impact.

1.15 Survival Aspects

The accident was survivable. The cabin area near rows 3 and 4 was damaged most severely; the floor and sidewalls were distorted. Only three of the occupied passenger seats remained intact and all of those seats showed some degree of distortion. All of the remaining occupied seats were either totally or partially separated from the sidewall attach points and legs. Typical failures involved the separation of the tubing from the seat frame support structure. The failures were usually located adjacent to the point at which the seat leg attachment bracket was welded to the seat frame support tubes. The failures were usually accompanied by the separation of one or both fuselage sidewall attachment points.

The captain's seat remained attached to the floor with the top of the seatback tilted to the right. The rear of the seat pan was cracked and the seat pan support tube and corner brace were separated from the vertical tubes of the seatback. The first officer's seat remained attached to the floor. The right corner brace was bent slightly.

The captain's feet were trapped in the wreckage. In addition, he suffered a severe back injury and had to be extricated by rescuers. The first officer also suffered a back injury but was able to escape through a broken cockpit window.

The passengers reported that the flight attendant's announcements reassured them and that there was no panic before or after the impact. Most of the passengers in the forward compartment were trapped under debris and other passengers. Only one man was able to free himself from the area. At least three passengers, who were seated in the rear of the cabin, were able to open the right rear window and escape without assistance.

Although four passengers reportedly took the brace position just before impact, the flight attendant did not specify or recommend the brace position as a preimpact, emergency procedure. The company had no procedure that required the flight attendant to instruct the passengers to assume a brace position.

The crash/fire/rescue (CFR) efforts were coordinated by the Port Authority of New York and New Jersey Airport CFR organization. The airport units arrived at the accident site within 2 minutes of the alarm; the fire station was about 1/2 mile away. Units from the Newark fire and police departments also responded with 15 CFR units and 12 ambulances.

1.16 Tests and Research

1.16.1 Metallurgical Examination of Black Tail Rotor Blade

The tail rotor blade was composed of a leading edge spar, honey-combed interior and an aluminum outer skin which was bonded to both the spar and the honeycomb.

Fatigue cracks were found through significant portions of both the leading edge spar and the bonded skin of the separated tail rotor blade. No defects which could have started the fatigue were found at either origin.

The black tail rotor blade separated about 15 in. from the butt end of the blade. An area of disturbed paint extended along a line from a point on the trailing edge at the fracture to a point near the leading edge at the tip of the blade. The inboard surface contained a similar, but smaller, area of disturbed paint. (See Figure 1.) The inboard portion of the blade had been damaged substantially after the outboard portion separated from it.

The fracture on the outboard portion of the blade was examined with the aid of a bench binocular microscope. Markings indicative of preexistent fatigue cracking were found in both the blade spar and skin. The cracking in the spar began at the outboard corner of the aft face of the spar. (See Figure 2.) The fatigue extended through 90 percent of the spar cross-section before the spar failed.

Fatigue in the skin appeared to initiate from multiple origins in an area on the bonded, interior surface of the skin. Fatigue propagated away from the origin area in opposite directions toward both the leading and trailing edges. The skin fatigue crack extended around the leading edge out to the area indicated by the arrow T in Figure 2. The skin fatigue in the other direction extended to a point about 1 1/2 in. from the leading edge. The total length of the skin fatigue crack was less than 2 in.

An area of skin at the fracture had been gouged on the outboard face of the blade near the leading edge. The gouge was about 0.2 in. long. (See Figure 2.) A portion of the bond material beneath the gouge had been squeezed out from between the skin and spar, indicating that the skin had been gouged after separation of the blade.

The area of the spar in which fatigue originated is shown in Figure 3. A dark stain was found in the immediate origin area. The most probable location of the origin would be at the center of this stain area as shown by the arrow in Figure 3.

The radius of the corner of the spar near the origin was about 0.009 to 0.011 in. The radius could not be measured precisely because of the uneven surface of the spar. The engineering drawing for the spar specifies that the radius at this location must be between 0.025 in. and 0.050 in. (See Figure 4.)

Figure 1. Overall view of the outboard face of the fractured tail rotor blade. (Arrows denote line of disturbed paint.)

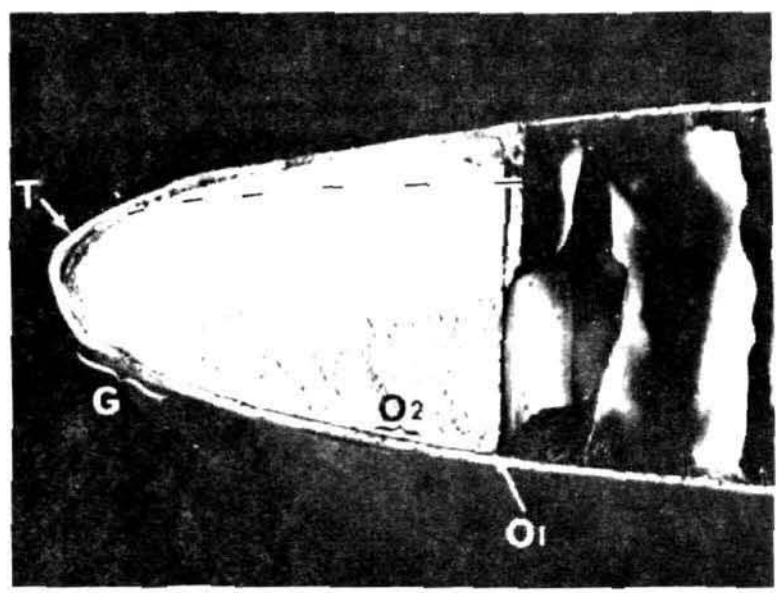


Figure 2. Fracture at the leading edge of the blade. Arrow "01" indicates the fatigue origin in the spar and bracket "02" the origin area in the skin. Arrow T indicates the terminus of the portion of the skin fatigue crack which propagated toward the leading edge. Bracket G indicates a gouge in the skin (X3 approximate).

Figure 3. High magnification visible light photograph of the origin of the fracture on the spar (arrow). X400

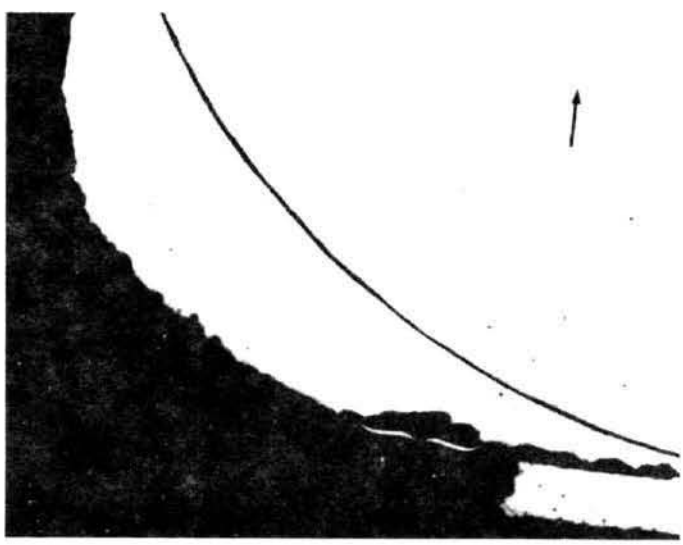


Figure 4. Spar section adjacent to the fracture. Arc represents a 0.025 in. radius. Arrow points in the direction of the leading edge. (X200, as polished.)

In order to determine if the radii on the aft surface of the spars on the remaining tail rotor blades met specifications, a section was cut from each of the blades at locations similar to the fracture location on the black blade. Only the outboard radius of the white blade spar appeared to be within the specified limits. The inboard radius on the leading edge spar on the white blade and both radii on the leading edge spars of the remaining three blades were irregular and below the 0.025 specified minimum radius.

The following is a list of the approximate material hardness values obtained from sections cut from the various spars:

Blade	Average Rockwell "E" hardness number[3/]
Black	93.5
Blue	94.5
Yellow	95
Red	94
White	91

1.16.2 Metallurgical Examination of the Tail Rotor Gearbox and Gearbox Mounting Plate

The tail rotor gearbox was bolted to the tail boom through four mounting flanges. Three of the gearbox mounting flanges had fractured and at the the fourth flange, the left aft position, the barrel nut had become disengaged from the bolt. All fractures in the mounting flanges of the gearbox appeared typical of overload separations. (See Figure 5.)

The bolt from the left aft flange position was bent slightly, and the threads were damaged extensively, particularly those threads closest to the bolt head. (See Figure 6.) Most of the thread appeared to have been damaged by transverse contact directly on the crowns of the threads, which resulted in widened crowns and compression bulging of the thread flanks. (See Figure 7.)

Two portions of one of the threads contained a different type of damage. The threads in these areas had been deformed in a direction away from the bolt head by heavy contact along the flank. One of these areas is indicated by bracket W in Figure 7. The second such area was on the same thread approximately 90° around the bolt.

3/ The ASM Metals Handbook, Volume 1, lists an average hardness of Rockwell "E" 85 to 97 for 6061-T6 aluminum (specified material for spar).

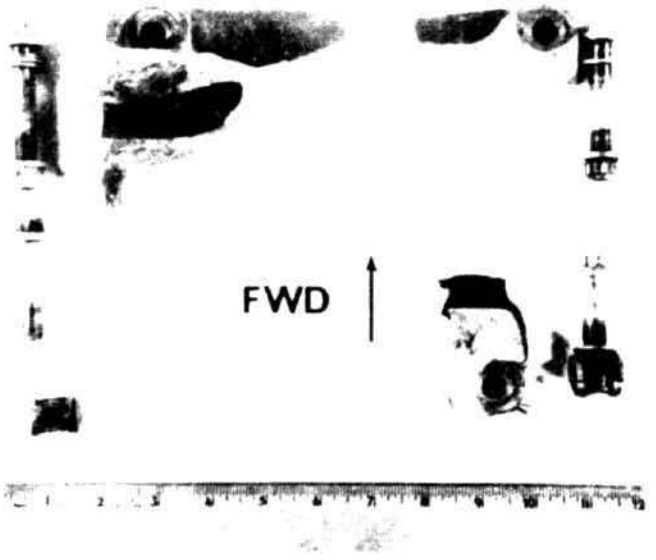


Figure 5. Tail rotor gearbox mounting bolts and nuts with fractured pieces of the gearbox mounting flanges.

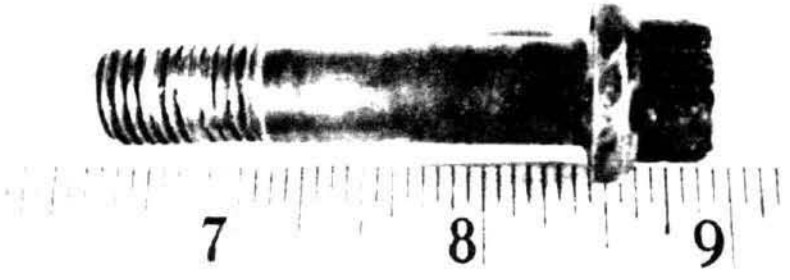


Figure 6. Left aft gearbox mounting bolt (lower left bolt in Figure 5). Note that this bolt in Figure 5 contains tape on the shank just below the head which was used to maintain identification.

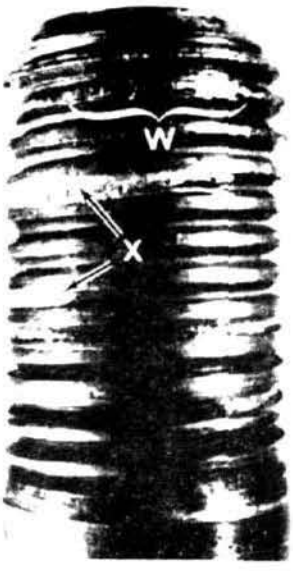


Figure 7. Thread damage on the left aft bolt. "W" denotes area containing deformation. "X" denotes areas where metal deposits were found. (X4)

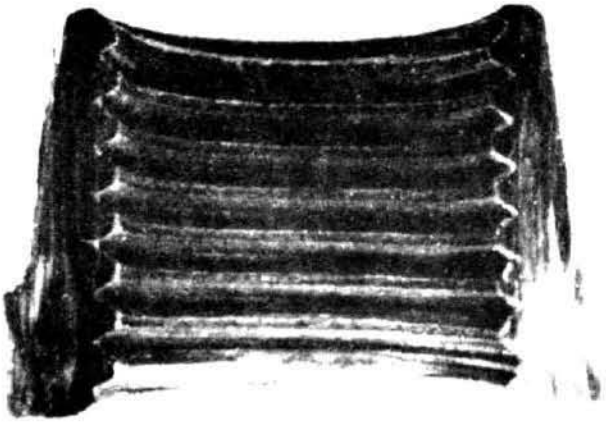


Figure 8. View of interior threads of the left aft barrel nut. (X4)

Foreign metal deposits were found in some areas between the threads. (See Arrow X Figure 7.) This material was rich in aluminum rather than iron, the bolt material. This indicated that the source of the foreign material was probably the aluminum gearbox mounting plate. About 225° of the thread root circumference closest to the bolt shank also contained shallow cracks.

Examination of the threads of the barrel nut from the left aft position of the gearbox mounting plate revealed only one damaged thread. This thread was the one which would be closest to the bolt head when assembled. The thread was deformed in the direction of the bolt head, apparently by hard contact with the thread flank. (See Figure 8.)

Both the left aft bolt holes in the gearbox and in the mounting plate had been deformed into an oval shape. (See Figure 9.) Most of the damage was on the left and right sides of the holes. The damage to these bolt holes appeared to have been produced by relative motion between the gearbox and mounting plate in the left and right directions while the bolt was in place.

1.16.3 Additional Testing of Black Rotor Blade

The outboard section of the blade was tested for evidence of bonding voids and to determine the bonding characteristics between the blade skin and the core. The blade was x-rayed, and coin tap and peel tests were conducted. No discrepancies were noted except in the areas immediately adjacent to the fractured surface.

In another test, a new tail rotor spar was notched intentionally about 14 in. from the blade attachment holes on the outboard spar. The blade assembly was assembled using the notched spar and subjected to a single combined edgewise/flatwise load level accelerated beyond a normal cruise level in order to initiate a fatigue fracture from the notch.

Once the crack was initiated, the blade assembly was subjected to a spectrum of loads which were similar to loads which would result from a New York Airways operation. The spectrum of loads was repeated every 3 aircraft hours. At the end of each 3-hour block, the blade was inspected ultrasonically. This procedure was repeated until a complete fracture occurred. The fracture occurred about 54 hours after crack initiation.

Additional testing of similar blade assemblies are underway. However, the current 6-hour inspection requirement imposed by the FAA for blade assemblies with more than 1,200 hours appear adequate based on the findings of the first test.

1.17 Other Information

1.17.1 Emergency Procedure For Tail Rotor Malfunction

The Sikorsky Aircraft S-61L Flight Manual Part 1, Section III, Emergency Procedures, states:

Figure 9. View looking forward of the left aft bolt holes in the tail rotor gearbox (bracket "1") and the gearbox mounting plate (bracket "2"). Also visible is the barrel nut hole (bracket "3"). (X2/3)

Tail Rotor Malfunction

The most probable type of tail rotor malfunction is a drive system malfunction whereby tail rotor rpm and thrust are lost. This may be caused by fracture of the shaft, coupling, or gearbox, or separation of the tail rotor assembly from the helicopter. Tail rotor separation from the helicopter is usually caused by severe vibration that has been induced by the fracture of a rotating component. A drive system malfunction is the most difficult type to cope with as it is accompanied by the loss of the rotating disc area that normally acts as stabilizing fin in forward flight. In most cases, extended flight is not possible after tail rotor malfunction and a sudden malfunction at high speeds may produce violent aircraft response. If excessive vibration or unusual noise is noted in the tail section during forward flight, airspeed should be immediately reduced to the best autorotational speed, as this may be an indication that a tail rotor malfunction is imminent. An early indication of a tail rotor malfunction is the loss of directional control. When a drive system malfunction in which loss of thrust occurs is experienced, the main rotor torque will yaw the helicopter to the right. The rate and amount of yaw will be governed by the amount of power applied and the airspeed at the time of the malfunction. The yaw tendency can only be reduced by an immediate reduction in power. Extended flight with a nose right yaw is not possible and it is recommended that a full autorotation be entered immediately upon detection of a tail rotor malfunction. Landing without tail rotor thrust is considered hazardous. Autorotation may occur with a nose right sideslip of greater than 45° and a greater rate-of-descent than normal autorotation. This will require a modified side flare to reduce the ground speed to as near zero as possible.

Tail Rotor Malfunction During Flight

Tail rotor malfunction which results in loss of tail rotor thrust during flight will be indicated by a loss of directional control accompanied by a rotation of the nose to the right. The first and most important step is to regain directional control by an immediate reduction of power to the main rotor. Do not attempt to extend flight by pulling power. Immediately perform the following procedures:

1. Reduce power to the main rotor and establish a glide at 65 to 75 knots IAS to regain directional control.
2. Choose the least hazardous landing site.
3. Maintain directional control by lateral movement of the cyclic control stick.
4. Alert passengers and fasten seatbelts.
5. Wheel brakes - OFF.
6. Tail wheel - LOCKED.
7. Make final approach into the wind if possible.
8. Shut down both engines.

APPROACH AND LANDING WITH TAIL ROTOR INOPERATIVE.

1. Accomplish moderate flare at approximately 100 feet to reduce rate-of-descent and airspeed by using aft cyclic.

2. Level off at 30 feet.

3. Collective pitch lever — Increase to cushion landing and touch down in a level attitude.

4. Collective pitch lever — Slowly reduce collective pitch to minimum after ground contact and move cyclic stick slightly forward.

5. As soon as helicopter is firmly on the ground, wheel brakes AS REQUIRED.

6. Secure helicopter.

1.17.2 Inspection Procedure

The tail rotor assembly and tail rotor blades were inspected by New York Airways, Inc., maintenance personnel in accordance with the procedures and at established time intervals.

Sikorsky Aircraft Service Bulletin (SB) 61B15-24B, Tail Rotor Blade Shank Reinforcement Brackets Inspection, required a visual inspection "prior

to 1st flight of each day or every 6 hours, whichever occurs first;" SB61B15-1D Tail Rotor Blade Inspection for Cracks, required a visual inspection "before 1st flight of each day and every 6 flight hours." The inspection required, in part:

(1) Clean blade with a clean, dry cloth.

(2) Sight along blade on both sides and inspect entire length of skin for cracks, paying special attention to the inboard 32 in. of the blade.

(3) If a crack is suspected, strip paint in area of suspected crack.

Warning: If a crack is found in a blade, remove the blade immediately.

On April 18, the tail rotor blades were inspected twice at New York Airway's facility at John F. Kennedy International Airport. The first inspection was made in the morning before the helicopter departed on its first flight. The second inspection was completed about 45 minutes before the accident. Each inspection was completed according to established procedures by two mechanics and a maintenance supervisor. The mechanic who conducted the second inspection stated that he wiped each blade individually. He paid particular attention to the white painted areas for cracks, dents, and bulges. He observed no defects.

1.17.3 Black Tail Rotor Blade History

The black tail rotor blade, part No. S6115-30001-044, serial No. 61V-10042-10842, was manufactured on April 28, 1977, and shipped to New York Airway's, Inc., on May 12, 1977. The blade had been installed and removed from two other New York Airway's, Inc., helicopters before being installed on the accident helicopter on September 23, 1978. The total time on the blade was 2,444:01 hrs since manufacture.

A review of the blade maintenance records did not reveal any indication of service difficulties, vibrations, or cracks. According to the manufacturer, the blade was the first S-61 tail rotor blade to fail under these circumstances.

1.18 New Investigation Techniques

None

2. ANALYSIS

Failure of the Black Tail Rotor Blade

The accident sequence was initiated by the failure of the 35-in. section of the black tail rotor blade. The blade broke off when Flight 972 was about 1 mile

south of the accident site. The loss of the blade caused the loud noises and severe vibrations reported by the crew. A progressive-type fatigue crack propagated across 90 percent of the cross-section of the leading edge spar. The spar fatigue originated at the outboard side of the aft face of the spar. The tail rotor blade failed as a result of preexisting fatigue cracks in the outboard edge spar and skin. The fatigue began in the spar before it started in the skin. (See Appendix G.)

The metallurgical examination of the spar disclosed that fatigue began in a radius which did not meet engineering drawing specifications. However, radii in similar locations on the other four tail rotor blades were of similar dimensions. Therefore, the improper radius alone did not initiate the fatigue crack in the spar. Since no one item could be singled out as the primary initiator of the fatigue crack, the Safety Board was not able to determine exactly what initiated the fatigue failure of the tail rotor blade.

The Safety Board attempted to estimate the number of cycles necessary for the skin to crack as the fatigue progressed from the origin to the terminus. Through the use of a scanning electron microscope, the number of cycles was determined to be about 150,000. The tail rotor blade would experience 150,000 cycles of stress in less than 2 hours of flight time if each rotation of the tail rotor blade introduced one cycle of stress. (The tail rotor operates at 1,243 rpm.) It is not known whether the fatigue propagated during all stages of flight.

Obtaining an estimate of the number of striations proved to be more difficult for the spar fatigue crack than for the skin fatigue crack. In many areas the striations were unresolvable or obscured, probably as a result of rubbing between the mating fracture surfaces. For these reasons an estimate of the number of striations in the spar was not made.

Initially, the spar crack may have propagated as a result of lower frequency stress cycles, such as startup/shutdown loads, or when the blade was exposed to higher stresses during maneuvers. In this case, the amount of flight time for total crack extension would be significantly more than 2 hours.

The provisions of the various Service Bulletins had been adhered to, and all of the required inspections had been made and documented by New York Airways, Inc. However, since the section of the spar which failed was completely enclosed in an aluminum envelope, a fatigue crack which had begun in the spar could not be seen during a visual inspection and would not necessarily distort the skin of the blade at a specified time before failure. The examination of the failed blade indicated that the fatigue crack in the skin was present when the second inspection was made, about 45 minutes before the accident. At that time the crack was less than 2 in. long. Since a typical fatigue crack is tight, the crack in the skin of the blade was probably difficult to see, especially since it was less than 2 in. long. Consequently, while inspection procedures were adequate for the detection of certain faults in a tail rotor blade, they did not assure detection of a skin fatigue crack which resulted from the fatigue failure of the spar.

Failure of the Tail Rotor Gearbox and Assembly

The failure of the rotor blade generated an unbalanced force in the rotor system which caused three of the four tail gearbox attachment lugs to fail under static overload. As a result, the tail rotor and tail rotor gearbox assemblies separated from the helicopter and complete directional control was lost.

The fourth tail gearbox attachment lug remained intact and in place on the tail gearbox housing. The lug hole and the corresponding hole in the gearbox attachment plate were elongated. The bolt was bent slightly, and the threads damaged badly. The barrel nut for the bolt was in place in the pylon and relatively undamaged. The lack of damage to the threads of the barrel nut indicated that the bolt had lost torque and partially backed out of the nut before the tail rotor gearbox and assembly separated from the tail pylon.

The damage to the threads of the left aft bolt could only have occurred if the bolt had at least partially backed out and if the gearbox was capable of motion relative to the mounting plate at the bolt hole. Relative motion between these components could occur only if another of the gearbox attach points had been broken. Therefore, the left aft bolt threads were damaged after the black tail rotor blade separated and at least one of the gearbox attachment lugs had been fractured. The Safety Board could not determine if the bolt had partially backed out before the failure of the tail rotor blade, or as a result of the failure of the blade.

Any adverse effects of a partially backed out attachment bolt would probably be shared equally by all five tail rotor blades. However, the investigation did not indicate any additional fatigue failures in the remaining blades. The Safety Board concludes that a loose tail rotor gearbox was not the cause of the fatigue initiation.

Flightcrew Decisionmaking

The flightcrew was certificated properly and was qualified for the flight. They had received the off-duty time required by regulation, and there was no evidence that medical or physiological factors might have affected their performances.

When the tail rotor failed, Flight 972 was at an altitude of about 1,200 ft, and 1 mile east of the airport over the congested Port Newark area. The crew recognized immediately that the tail rotor system had malfunctioned. As a result, they had two options - - either to return to the airport or to autorotate. They chose to attempt an emergency landing at the airport for several reasons: (1) The airport, which was the only suitable forced-landing area, was less than a mile away. (2) Once the appropriate emergency checklist had been accomplished and the helicopter had been slowed to between 60 and 70 kns, the crew was able to control the helicopter in spite of the severe vibrations. (3) There was no yaw to indicate a drive shaft malfunction and resultant loss of tail rotor rpm. (4) They knew that if autorotation was initiated the rotor rpm would increase to about 105 to 108 percent, which would place a greater load on the tail rotor system. (5) The

first officer believed that the additional rpm would have increased the vibration and caused complete loss of control of the helicopter. (6) There was no suitable landing area below them.

Autorotation may have resulted in an earlier landing, but it may also have caused the gearbox to separate sooner because of increased rotor rpm's and increased stress on the gearbox mounting flanges. Also, it probably would have resulted in a landing in a hazardous area of Port Newark. Therefore, because extended flight was not necessary, the helicopter was not yawing significantly, and the flightcrew had no reason to believe that the gearbox mounting flanges would fail in the short period of time needed to reach the airport, the Safety Board concludes that the flightcrew's decision to make a controlled emergency landing at autorotational speed was an appropriate decision.

When the tail rotor gearbox and assembly separated from the tail pylon, the loss of the 180-lb unit caused the center of gravity to change immediately to 260 in., or within 2 in. of the forward limit, and resulted in an abrupt nosedown change in pitch. Simultaneously, directional control was lost when the tail rotor fell off. The helicopter began an immediate descending right turn of about 270° before hitting the ground. Although all directional control was lost, had the aircraft been at a higher altitude, it might have been flared sufficiently to cushion the touchdown on the airport; however, touchdown still would have been accomplished from a spiraling turn. Although the first officer did attempt to slow the rate of descent by initiating a flare before impact, the nosedown attitude and the low altitude of the helicopter prevented him from raising the nose high enough to cushion the touchdown.

Survivability

The accident was partially survivable because the g forces were within the range of human tolerance and because of the minimal disruption of the fuselage structure and minimal reduction in the occupiable volume of the fuselage. (See Figure 10.) The fatalities and severe injuries were caused by failures of passenger restraint systems under comparatively high vertical g forces in the forward portion of the helicopter.

The vertical g loads, which exceeded the 4-g minimum certification standard imposed for helicopter seats by Civil Aeronautics Manual 7.260, were variable, but they probably were in the range of 15 g's in the forward cabin. This estimate is based on the damage to the helicopter. In the 15-g range of impact force, the U.S. Army Crash Survival Design Guide, TR 71-22, October 1971, indicates that only moderate injuries should be expected when adequately restrained persons encounter impact forces of this magnitude.

The impact forces caused the sidewall tiedowns to fail and separate, which caused many of the seats to separate or become loose. The sidewall tiedowns probably separated when the fuselage sidewalls flexed on impact. Controlled crash testing of various aircraft and helicopters conducted by the National Aeronautics and Space Administration indicate that fuselage sidewalls flex several inches in crashes similar to that experienced by Flight 972. As a result, seat pan attachments will separate from the fuselage. Once the seat pan has separated, it

will swing through a wide arc until the passengers come into contact with other objects. Extensive sidewall flexion was probable in the forward fuselage of Flight 972 because the landing gear lateral supports were attached to the outside of the fuselage in this area. The forces on the landing gear would have been transmitted directly to the fuselage walls and flooring, causing localized flexion of the walls and distortion of the floor. As a result, the passengers in the forward cabin were probably thrown when the sidewall tiedown structure failed.

The Safety Board is aware that similar inadequacies in design requirements for passenger seats for general aviation aircraft exist. Therefore, improvements in general aviation aircraft crashworthiness have been made a special safety objective. We expect to include improvements in design requirements for helicopter passenger seats in this objective.

Four occupants reported taking a brace position before the initial impact. These occupants received less severe upper torso and head injuries than other occupants seated in the same rows who did not assume a brace position. For example, one passenger who was in the brace postion received only minimal injuries although both occupants seated beside him were killed. Another passenger, a former U.S. Army helicopter crewman, assumed a brace position and received only minimal head and upper torso injuries while the passenger next to him received a serious open, depressed frontal skull fracture.

The emergency procedures and the passenger briefing cards should have specifically required the flight attendant to instruct passengers to assume the standard brace position, which would have reduced the possibility of serious injuries during the emergency landing.

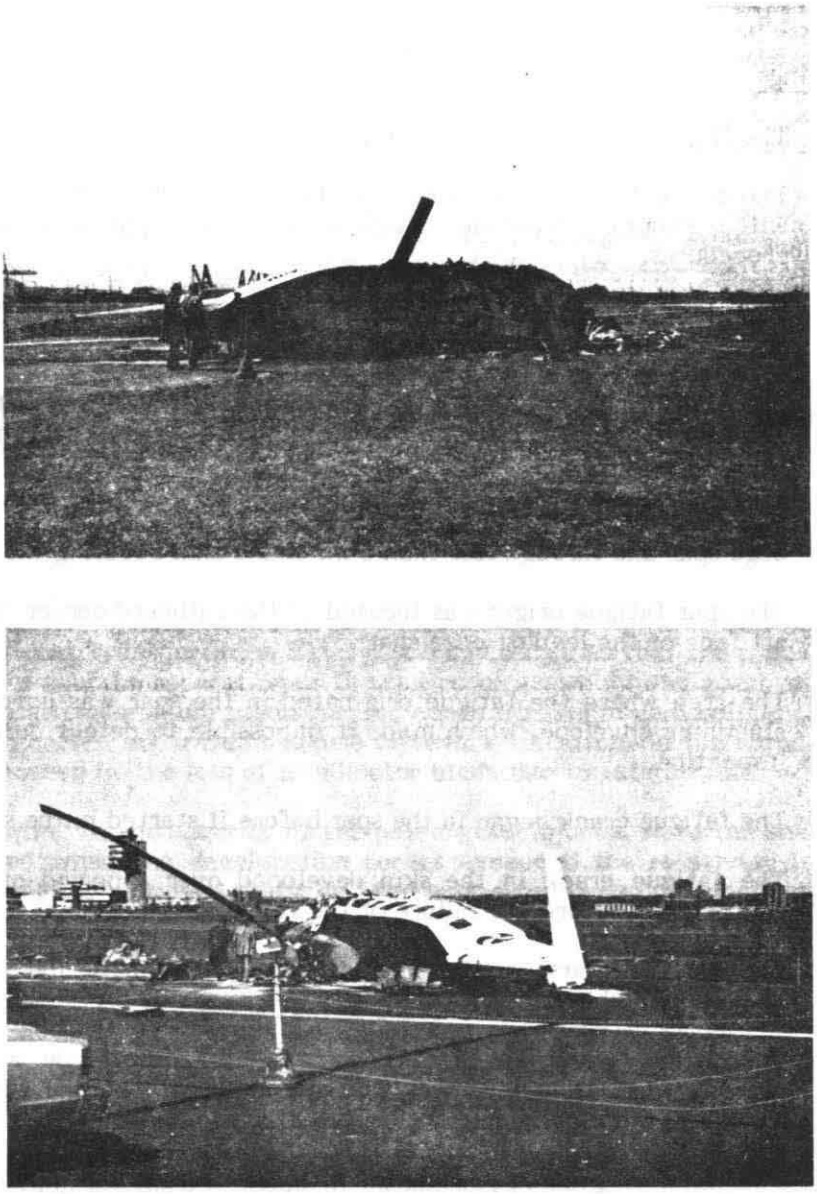


Figure 10. Views of structural damage.

3. CONCLUSIONS

3.1 Findings

1. The flightcrew was certificated properly and qualified for the flight.

2. The aircraft was certificated, maintained, and dispatched in accordance with Federal Aviation Regulations and approved maintenance procedures.

3. The black tail rotor blade broke when the helicopter was at 1,200 ft and about 1 mile east of the airport.

4. The blade broke because it had been weakened by a preexisting fatigue crack in the leading edge spar and the blade skin about 35 in. from the tip of the blade.

5. The fatigue crack had propagated through 90 percent of the leading edge spar and through less than 2 in. of the skin covering the spar.

6. The spar fatigue origin was located at the outboard corner radius of the aft face of the leading edge spar.

7. The area where the fatigue originated in the spar was covered by the aluminum envelope, which made it impossible to detect during a visual inspection.

8. The fatigue crack began in the spar before it started in the skin.

9. The fatigue crack in the skin developed over a period of at least 2 flight-hours before the failure of the tail rotor blade.

10. The fatigue crack in the spar developed over a period significantly longer than 2 hours.

11. The fatigue crack in the skin of the tail rotor blade was probably present when the blade was inspected 45 min. before the accident. However, the crack could not be detected readily.

12. The visual inspection procedures in effect were not adequate to detect the fatigue crack in the skin which developed from a fatigue crack in the spar.

13. The loss of the 35-in. section of the tail rotor blade did not cause a loss of directional control.

14. The loss of the 35-in. section of the tail rotor blade generated unbalanced forces, which caused three of the four tail rotor gearbox housing attachment flanges to break from static overload and a portion of a thread of the bolt in fourth flange to fail.

15. The failure of the attachment flanges permitted the gearbox and assembly to separate from the tail pylon, which caused the loss of directional control and an abrupt nosedown pitch attitude change.

16. The helicopter's low altitude when directional control was lost and the nosedown pitch change made further control of the helicopter impossible.

17. The flightcrew's decision to attempt an emergency landing at Newark International Airport was an appropriate decision under the circumstances.

18. The accident was partially survivable.

19. The failure of the seat support and tiedown structure contributed to the number of fatalities and serious injuries.

20. Fewer serious injuries would have occurred if the passengers had taken a "brace" position.

3.2 Probable Cause

The National Transportation Safety Board determines that the probable cause of this accident was the separation of the tail rotor assembly and gearbox from the aircraft at an altitude which made further controlled flight impossible. The rotor assembly and gearbox separated because of severe vibrations in the rotor assembly which were induced by the loss of a tail rotor blade due to fatigue failure.

Contributing to the severity of the passengers' injuries were the seat failures which occurred when the deceleration forces exceeded the relatively low design strength of the FAA-approved seats, and the lack of guidance on a passenger brace position for emergency landings.

4. SAFETY RECOMMENDATIONS

On April 19, 1979, the National Transportation Safety Board issued the following safety recommendations to the Federal Aviation Administration:

> Withdraw the airworthiness certificates of Sikorsky S61 helicopters until a means of detecting potential tail rotor blade failures can be devised and implemented. (Class I—Urgent Action) (A-79-25)
>
> Notify foreign operators of Sikorsky S61 aircraft of this action. (Class I—Urgent Action) (A-79-26)

The FAA, in response to these recommendations, issued an Airworthiness Directive on April 20, 1979, requiring S-61 operators with PN 6115-30001 and 6117-30001 series tail rotor blades installed to (1) perform a one-time dye penetrant inspection of the inboard 32-in. section of each blade and of the tail rotor gearbox housing attachment lugs before further flight, (2) conduct a visual inspection of

each blade skin every 6 flight-hours for evidence of cracks, (3) ultrasonically inspect each blade spar and skin for cracks within 6 flight-hours, and (4) repeat the ultrasonic inspections every 6 flight-hours on tail blades which have accumulated 1,200 or more flight-hours.

Results of the initial ultrasonic inspections revealed no additional spar cracks. The metallurgical examination of the failed tail blade and skin did not disclose any abnormalities in the material used or the manufacturing techniques. The manufacturer is conducting additional fatigue testing of "notched" tail blade spars to determine propagation rates. The initial results of these tests indicate that the 6-hour ultrasonic inspection interval required by AD is adequate to detect potential tail rotor blade defects.

Also as a result of this investigation the Safety Board has issued the following recommendations to the Federal Aviation Administration:

> "Establish a research project to determine the optimal brace position for various seat designs and seating configurations on aircraft used in passenger-carrying operations. (Class II, Priority Action) (A-79-76)
>
> "Issue an Air Carrier Operations Bulletin on the basis of this study requesting principal operations inspectors to insure that the training of crewmembers includes information on the appropriate passenger brace position for specific aircraft configurations during potential crash landings. (Class II, Priority Action) (A-79-77)
>
> "Issue an Air Carrier Operations Bulletin requiring principal operations inspectors to instruct their assigned air carriers to describe the appropriate emergency brace position on the passenger briefing card and to require that preflight briefings include a reference to the proper brace position. (Class II, Priority Action) (A-79-78)"

ICAO Note: Appendices A, B, C, D, E and F, of the Report are not reproduced.

ICAO Ref.: 120/79

Appendix G

Metallurgical Information

A single, fatigue origin was found on the spar, indicating rather low initiating stresses since no significant stress risers were found. The fatigue initiation area of the skin contained a large number of origins, indicating higher initiation stresses. Thus, fatigue cracking in the spar probably resulted in higher loads being transferred into the skin.

The first 3/8 in. of the spar fracture was darker in appearance (under visible light) than the remaining fatigue fracture in both the spar and skin. The darker appearance in general indicated older age because of longer time to weather or mechanical rub, or both.

The location of the skin origin area was about halfway between the spar origin and the terminus of the darker area of the spar. This is consistent with propagation of the spar fatigue followed by initiation of the skin fatigue from multiple origins because of the load transferred to the skin from the cracked portion of the spar.

The fracture surface in the vicinity of the spar origin was extremely smooth and few striations were evident. The fracture surface in the vicinity of the skin origin area was rougher and individual striations were evident. This indicated that the propagation rate and propagation stresses were greater at the skin origin than at the spar origin.

No. 6

Britten Norman Islander BN-2A-9, DQ-FBO, accident near Bua, Vanua Levu, Fiji, on 12 July 1979.
Report No. 79-0-405 dated 7 January 1980 released by the Fiji Government.

SYNOPSIS

DQ-FBO departed from Nausori airport, Suva, for Bua aerodrome on the west coast of Vanua Levu at 0946 hours on 12 July 1979. The flight failed to arrive at Bua and the aircraft wreckage was located in the ranges east of Bua about 1313 hours on 14 July 1979. The nine occupants of the aircraft had all been fatally injured in the accident.

1. FACTUAL INFORMATION

1.1 History of flight

1.1.1 The pilot flew DQ-FBO on Fiji Air flights PC31 and PC32 from Nausori to Ba and return on a visual flight rules (VFR) local flight plan. Flight PC31 departed from Nausori at 0758 hours Fiji standard time* and arrived at Ba at 0836 hours. The aircraft departed from Ba 12 minutes later at 0848 hours and arrived at Nausori at 0926 hours.

1.1.2 While the aircraft was being refuelled and loaded the pilot went to the control tower to obtain the latest weather information before departing on Fiji Air flight PC71 from Nausori to Bua. The fuel load carried was 202kg (447 lbs) which was the company standard minimum fuel for a return flight from Nausori to Bua and provided a fuel endurance of 3 hours. The flight was operating on a standard VFR local flight plan and commenced taxiing at 0942 hours with 8 passengers on board.

1.1.3 DQ-FBO became airborne on runway 10 at 0946 hours. At 0949 hours the pilot advised Nausori air traffic control (ATC) "Bravo oscar departed 49." ATC replied, "Bravo oscar airborne time 46, report arrival Bua". The pilot acknowledged this message. No further communication was received from the aircraft.

1.1.4 Bua aerodrome is 71 nautical miles north of Nausori airport on a track of 003°T (350°M). The standard flight plan time from Nausori to Bua was 35 minutes and when no message had been received from DQ-FBO by 1023 hours Nausori ATC endeavoured to contact the aircraft on both very high frequency (VHF) and high frequency (HF) channels. In addition aircraft operating in the area north of Nausori were asked to call DQ-FBO.

1.1.5 An Air Pacific flight enroute from Nausori to Labasa at 7000 feet on an instrument flight rules (IFR) local flight plan descended to the west of track below cloud and proceeded to Bua. The flight was overhead Bua airstrip at about 1120 hours and the pilots observed that DQ-FBO was not on the ground at Bua.

1.1.6 The rescue co-ordination centre at Nadi was activated at 1215 hours and an extensive air and ground search was initiated.

1.1.7 The wreckage of the aircraft was located by a ground party about 1315 hours on 14 July 1979 5½ nautical miles east-north-east of Bua airstrip at an elevation of 1450 feet AMSL. The accident site was in steep terrain covered with tropical rain forest.

1.1.8 The accident occurred in daylight at about 1020 hours on 12 July 1979. The accident site was at 16° 49′ 21″ S. latitude, 178° 42′ 45″ E. longitude.

1.2 Injuries to persons

1.2.1	Crew	Passengers	Others
Fatal	1	8	—
Serious	—	—	—
Minor/none ...	—	—	—

*All times in this report are Fiji standard time.

1.3 Damage to aircraft

1.3.1 The aircraft was destroyed by impact forces.

1.4 Damage to property

1.4.1 **Nil.**

1.5 Personnel information

1.5.1 Pilot in command held Australian commercial pilot licence No. 116515 issued on 23 September 1976 and valid until 31 August 1979. Single engine group endorsements and twin engine endorsements for PA23 and BN-2 aircraft were entered on the licence. He held also a class one instrument rating and a C grade instructor rating both valid to 31 October 1979.

1.5.2 On 27 June 1979 the Fiji Department of Civil Aviation issued a certificate of validation No. 92 valid until 26 July 1979, authorising Captain "to exercise the privileges" of his commercial pilot licence whilst flying aircraft operated by Fiji Air Limited.

1.5.3 Captain's total flying experience was 1074 hours including 139 hours on BN-2 aircraft and 65 hours actual instrument flight time. During the 90 days preceding the accident he had flown 90 hours including 26 hours on BN-2 aircraft and 41 hours instrument flight time.

1.5.4 He commenced route training with Fiji Air Limited on 28 June 1979 and up to 5 July 1979 had completed 15 hours 20 minutes command training under supervision. This training included route flying on 27 sectors under VFR conditions with circuits and landings at Bua and 9 other airstrips. On 11 July he flew 2 hours 8 minutes under supervision on 4 sectors in VFR conditions with circuits and landings at 2 other airstrips. He had received no training in the IFR procedures applicable to routes operated by Fiji Air Limited.

1.5.5 He was authorised as a pilot in command with Fiji Air Limited on 5 July 1979 and at the time of the accident had flown 13 hours with the Companv as a pilot in command,

1.5.6 During the 7 days prior to the accident he had flown for 14 hours 26 minutes and had been on duty for 36 hours 25 minutes. He had a rest period of 12 hours 55 minutes prior to reporting for duty on 12 July 1979.

1.5.7 The pilot in command had not flown in Fiji prior to the 28 June 1979. He had flown the Nausori-Bua route twice prior to the accident, once under supervision and once as pilot in command. During the flight to Bua under supervision the check captain advised Captain that the approach to Bua was normally made by tracking seaward of the coast. The check captain also drew his attention to the need for caution when approaching Bua, if to the east of track, in view of the terrain immediately to the east.

1.5.8 The Operator had instructed Captain verbally not to fly in Instrument Flight Conditions until he had passed an instrument rating flight test in Fiji. In addition the Operator "had stressed that the pilot in command was not expected to push on into bad weather and he was not to hesitate to abort a flight if he was at all unhappy with the conditions".

1.6 Aircraft information

1.6.1 Britten Norman BN-2A-9 Islander aircraft, DQ-FBO, Serial No. 195 was manufactured in 1970 and had flown a total time of 8835 hours and 510 hours since the last Certificate of Airworthiness renewal inspection.

1.6.2 Two Lycoming 0-540-E4CB engines were installed. No. 1 engine, Serial No. L-18977-40, had run 2154 hours since new. No. 2 engine, Serial No. L-18164-40, had run 2182 hours since new and 190 hours since overhaul. Hartzell constant speed full feathering propellers type HC-C2YK-2C/8477A-4 were fitted. No. 1 propeller, Serial No. AU490 had run 6631 hours since new and 1289 hours since overhaul. No. 2 propeller, Serial No. AU5372 had run 1352 hours since new.

1.6.3 The aircraft was fitted with dual flying controls and was instrumented and equipped in accordance with current regulations for IFR operation. It was equipped with one Narco mk 12A VHF communications transceiver and a VHF Omni-Directional Radio Range (VOR) receiver. One AWA HF communications transceiver was also installed. Two Bendix T12C medium frequency automatic direction finding (ADF) receivers were fitted.

1.6.4 Integral wing tip fuel tanks with a usuable capacity of 104 litres (23 Imp gallons) per side were installed. The fuel from each tip tank was led to an electrically actuated two-way fuel cock interposed between the inner wing tank and the auxiliary fuel pumps. These fuel cocks were actuated by switches mounted on an auxiliary panel fitted to the windscreen dividing panel.

1.6.5 All mandatory modifications required to the airframe, engines, propellers and equipment had been embodied and all required special inspections had been carried out.

1.6.6 The Fiji Certificate of Airworthiness No. 53 in the Transport Category Passenger had last been renewed on 25 March 1979 and was valid until 24 March 1980.

1.6.7 The Certificate of Maintenance No. 676 had been issued on 4 July 1979 and was valid to 18th August 1979 or attainment of 8850 flying hours whichever occurred the earlier.

1.6.8 The total fuel on board DQ-FBO at the time of departure from Nausori was 202 kg (447 lbs) of 100/130 octane Avgas. Both tip tanks were full (74 kg each) and there was 27 kg of fuel in each main tank. All the aircraft equipment was serviceable with no recorded defects.

1.6.9 The right hand front seat next to the pilot was not occupied on the accident flight.

1.6.10 The maximum zero fuel weight was 2766 kg (6100 lbs) and the maximum take-off and landing weights were 2857 kg (6300 lbs). The load-sheet for the flight recorded a zero fuel weight of 2550 kg (5622 lbs) and a take-off weight of 2752 kg (6069 lbs). The passengers' actual weight had been used in compiling the loadsheet. Although the centre of gravity (C of G) position was recorded as "WIL" (within limits) no calculation had been made to ascertain the actual C of G.

1.6.11 During the investigation following the accident the C of G was calculated to have been 65.76 cm (25.893 inches) aft of the datum point at the wing leading edge. The aft C of G limit was specified as 65.02 cm (25.6 inches) aft of the datum point.

1.7 Meteorological information

1.7.1 On the morning of 12 July 1979 Fiji lay in a broad area of light variable winds between anticyclonic systems to the east and west, the equatorial trough well to the north and a depression to the south near northern New Zealand. Several bands of cloud, oriented northwest-southeast, were drifting slowly eastward across the Fiji area.

1.7.2 A copy of the Fiji area forecast, valid from 0500 hours to 1200 hours on 12 July 1979 and the terminal area forecasts for Nausori and Nadi up to 1800 hours were issued to the pilot prior to departure from Nausori.

1.7.3 The area forecast was for scattered to broken stratocumulus (Sc) and cumulus (Cu) cloud with a base between 2500 feet and 5000 feet and visibility 35 km reducing to 8 km in rain showers. The wind velocity at 3000 feet was forecast to be variable at 10 knots.

1.7.4 The actual weather at Nausori when flight PC71 departed for Bua was overcast with a main cloud base of 5000 feet, $\frac{1}{8}$ Cu cloud at 2500 feet, visibility 60 km, temperature 23°C and dew point 21°C. ATC gave the surface wind velocity as 270°M less than 5 knots and the QNH as 1015 mbs. The Nausori terminal area forecast indicated that conditions would remain similar with the cloud becoming temporarily 4/8 Cu at 2000 feet in rain showers.

1.7.5 During the day, as an anticyclone over eastern Australia advanced eastward, westerly or southwesterly winds developed over the Fiji island group. The upper winds at Nadi measured from a balloon released at 1115 hours, were 225°T, 12 knots at 1000 feet increasing to 225°T, 18 knots at 3000 feet. The main low cloud over Fiji was a broad band of Sc at a height estimated at most stations as around 4000 to 6000 feet, with scattered Cu beneath at 2000 feet or higher but reported as between 1000 and 2000 feet at Savu Savu and Labasa. A few scattered showers, generally light, were reported during the day but visibilities were in the range of 20 to 50 km or better.

1.7.6 There was no weather reporting station at Bua. The nearest reporting station was at Nabouwalu, on the coast 12 miles south of Bua. At 0600 hours Nabouwalu reported 4/8 of Cu and Sc cloud between 2000 and 3000 feet with an overcast above. Visibility was given as 50 km and the surface wind was calm. At 0900 hours the cloud conditions were unchanged but visibility was stated as 20 km with light showers and the surface wind was variable at 2 knots. These actual reports were not available to the pilot before flight PC71 departed from Nausori.

1.7.7 The pilot of an aircraft which departed from Nausori at 0935 hours reported that as he proceeded north along the coast of Viti Levu to Tailevu Point conditions were showery and Ovalau Island was totally obscured in heavy rain-showers. At 1000 hours the aircraft was at Viti Levu Bay some 35 nautical miles south-southwest of Bua. Heavy rain shower activity could be seen about 20 miles to the north-east and the island of Vanua Levu was not visible. Thirty minutes later the aircraft was 10 miles north of Ba at an altitude of 3000 feet. The pilot stated that "the wind in this area was quite strong, blowing at 25 to 30 knots from 220°M".

1.7.8 A Fiji Air Limited aircraft departed from Nadi at 1002 hours and proceeded VFR at an altitude of 1500 feet via the south coast of Viti Levu to

Pacific Harbour. After a brief stopover the flight continued VFR to Nausori where it landed at 1102 hours. The pilot reported that he had experienced a strong south-westerly wind from Nadi until he reached the vicinity of Nausori, where the wind became light and variable.

1.7.9 The two caretakers employed at Bua aerodrome stated that when they left Ndama village 1½ miles south-east of Bua before seven o'clock, " it was just coming on to rain ". When they arrived at the aerodrome about seven o'clock " it was raining heavily and was very dark ". Steady rain fell until after ten o'clock. Some time after ten o'clock they heard the sound of an aircraft for a very short time to the south-east of the airstrip. They could not see the aircraft as " it was foggy ". The wind was blowing from the south-east at about 20 knots.

1.7.10 Two men who were on the top of a ridge at an elevation of about 1000 feet and 4 miles east-north east of Bua aerodrome heard the sound of an aircraft approaching from the south and flying toward the hills. The aircraft passed very close to them and was " not very high " or " about on the same level" as where they were standing. They did not see the aircraft as they were in cloud and light rain was falling. A short time later they heard a bang followed by sounds of rushing air before the noise ceased. They were not sure what the time was when this occurred.

1.7.11 The Air Pacific flight which diverted to Bua (see paragraph 1.1.5) descended in the Wainunu Bay area, some 15 miles east-south-east of Bua at about 1115 hours. The cloud base was approximately 4700 feet and the pilot observed that there was a line of stratiform cloud lying northwest-southeast across western Vanua Levu. The base of this cloud was not well defined and light rain and drizzle was falling from it. Flight visibility was more than 30 km to the west and 15 km to the north and east with hazy conditions over the hills and remnants of stratus in the valleys. The surface wind was a light south westerly. After circling Bua at about 1120 hours the flight made a search along the hills at an altitude of 2000 feet to the south to Nabouwalu then proceeded north again inland from Bua and thence north-eastward. The flight experienced light turbulance which became moderate over the ranges to the north-east of Bua. While making an instrument let-down to Labasa between 1150 and 1200 hours a strong south westerly wind was evident during the let-down procedure.

1.8 Aids to navigation

1.8.1 Nausori airport was equipped with a VOR transmitter operating on 112.2 MHz, a distance measuring equipment (DME) beacon operating on channel 59X and a medium frequency non-directional beacon (NDB) operating on 307 kHz. These aids were all serviceable when DQ-FBO departed from Nausori for Bua and at the time of the accident.

1.8.2 No aids to navigation were installed at Bua aerodrome.

1.9 Communications

1.9.1 Communications facilities at Nausori airport consisted of one VHF channel on 119.7 MHz and two HF channels operating on 6533 kHz and 3460 kHz. All these channels were serviceable on the day of the accident.

1.9.2 No communications facilities were installed at Bua aerodrome.

1.10 Aerodrome information

1.10.1 Bua aerodrome is situated on the coast at Bua Bay at an elevation of 60 feet AMSL. It consists of a gravel runway 1000 m long and 18 m wide aligned on a bearing of 298°/118°T. Two windsocks are installed. The only visual aids are white markers at the runway ends and adjacent to the runway sides.

1.11 Flight recorders

1.11.1 No flight recorder was installed in the aircraft nor was one required to be installed.

1.12 Wreckage and impact information

1.12.1 The aircraft wreckage lay in a heavily timbered gully beside a spur on the east side of a large valley. The entire area was covered with tropical rain forest, the forest canopy at the accident site being generally 25m (80 feet) above ground level.

1.12.2 After breaking off several small tree branches the aircraft struck two substantial trees about 8m (24 feet) above the surface. The damage to the port wing indicated that it had impacted heavily against one tree outboard of the port engine causing the outer wing to deflect forward and fracture at that point. This outer wing section had struck the other tree and the wing tip, including the fuel tank, had separated from the wing as it was severed by the second tree. The nose of the aircraft had then yawed to the left as the remaining inboard section of the port wing separated rearward from the fuselage at the wing root. The port propeller had parted from the port engine and the engine had separated from the wing. The aircraft then demolished a number of smaller trees which caused the starboard wing to separate just outboard of the starboard engine. The fuselage and the wing centre section with the starboard engine and propeller attached, came to rest approximately 50 m (164 feet) from the initial impact point.

1.12.3 Both wing tip fuel tanks had ruptured and separated from the wing structure, the port tank being some 20 m along the wreckage trail while the starboard tank had come to rest adjacent to the starboard engine. Both electric actuators for the tip tank fuel cocks were in the "open" position.

1.12.4 The pre-impact integrity of the flight controls in the cockpit area was established. The elevator control push-pull rod had fractured in overload 35 cm (14 inches) aft of the base of the control column. In the tail section the elevator drive lever had separated from the aft end of the elevator push-pull rod also due to an overload failure. The upper section of the fin and rudder was folded over on to the port tailplane. The elevator complete with its trim tab had separated from the tailplane. The extension of the elevator trim tab screw jack was measured at the accident site.

1.12.5 The flaps and ailerons had separated from the wings after impact. The position of the transverse operating rod from the flap actuator was consistent with the flaps being in the fully up (6 degree) position. Fractures observed in the aileron control cables were overload failures.

1.12.6 The twin heading bars on the Magnesyn compass indicator were set on 352°. However, the heading indicator needle was missing and there were no marks on the dial of the instrument to show what the position of the needle may have been at impact.

1.12.7 The millibar sub-scale on the altimeter from the pilots instrument panel was set at 1015 mbs and the sub-scale on the second altimeter was set at 1013 mbs. The mechanism of both altimeters was broken.

1.12.8 The microphone selector on the audio panel was in the " HF " position while the HF selector switch was " off ". However the switch toggle had been damaged so the " off " indication was considered to be unreliable. The ADF control boxes and ADF indicators were not identified in the wreckage.

1.12.9 The aircraft clock had stoped at seventeen minutes past ten.

1.12.10 The damage to the blades of both propellers was typical of an impact with power on the engines.

1.12.11 The wings of the aircraft had been approximately level at first impact with the trees and from this point the trajectory of the fuselage was horizontal and on a track of 006°T.

1.13 Medical and pathological

1.13.1 The pilot in command had undergone a full medical examination including a chest x-ray, electrocardiogram and audiogram examinations on 28 June 1979 and had been assessed as medically fit to hold a commercial pilot licence.

1.13.2 An autopsy and toxicology analyses were carried out and revealed no abnormalities. There was no evidence to suggest or show that the pilot in command was not in a normal state of health when he departed on the accident flight.

1.14 Fire

1.14.1 Fire did not occur.

1.15 Survival aspects

1.15.1 The accident was not survivable as the cabin area of the aircraft was virtually destroyed.

1.15.2 No Emergency Locator Transmitter was installed in the aircraft nor was one required to be installed.

1.15.3 The aircraft had penetrated beneath the forest canopy without leaving any marks on the trees which would have been readily discernible from the air. Intensive searches were conducted in the hills east of Bua by both fixed wing aircraft and helicopters but the wreckage was finally located by a ground party.

1.16 Tests and research

1.16.1 A flight test was conducted with an Islander aircraft at an all up weight of 2736 kg (6032 lbs) and the C of G at 58.9 cm (23.2 inches) aft of the datum. The elevator trim handwheel was set to a position at which the extension of the elevator trim tab screw jack was the same as that measured on DQ-FBO at the accident site. At this elevator trim setting the " in trim " airspeed was found to be 132 knots.

1.17 Additional information

1.17.1 Fiji Air Limited held approval from the Fiji Air Transport Licensing Board to operate two scheduled air services per week on the Nausori-Bua-Nausori route.

1.17.2 The following sections of Article 22 of the Colonial Air Navigation Order 1961 specified the requirements which were relevant to the route qualifications of the pilot of DQ-FBO :

(5) (a) The pilot designated as commander of the aircraft for the flight shall . . . . -

(i) have demonstrated to the satisfaction of the operator that he has adequate knowledge of the route to be taken, the aerodromes of take-off and landing, and any alternate aerodromes, including in particular his knowledge of—

the terrain,

the seasonal meteorological conditions,

the meteorological, communications, and air traffic facilities, services and procedures,

the search and rescue procedures, and the navigational facilities,

relevant to the route ;

(ii) . . . . . ;

(iii) have carried out as pilot in command not less than three take-offs and three landings in aircraft of the type to be used on the flight.

(b) In determining whether a pilot's knowledge of the matters referred to in sub-paragraph (a) (i) is sufficient to render him competent to perform the duties of aircraft commander on the flight, the operator shall take into account the pilot's flying experience in conjunction with the following—

(i) . . . . . ;

(ii) the influence of terrain and obstructions on departure and approach procedures at the aerodromes of take-off and intended landing and at alternate aerodromes ;

(iii) the similarity of the instrument approach procedures and let-down aids to those with which the pilot is familiar ;

(iv) the dimensions of runways which may be used in the course of the flight in relation to the performance limits of aircraft of the type to be used on the flight ;

(v) the reliability of meteorological forecasts and the probability of difficult meteorological conditions in the areas to be traversed ;

(vi) the adequacy of the information available regarding the aerodrome of intended landing and any alternate aerodromes ;

(vii) the nature of air traffic control procedures and familiarity of the pilot with such procedures ;

(viii) the influence of terrain on route conditions and the extent of the assistance obtainable en route from navigational aids and air-to-ground communication facilities ;

(ix) the extent to which it is possible for the pilot to become familiar with unusual aerodrome procedures and features of the route by means of ground instruction and training devices.

1.17.3 The rules applicable to VFR flight outside controlled airspace were specified in the Colonial Air Navigation Order 1961 as amended by the Colonial Air Navigation (Amendment) Order 1968 :

" An aircraft flying outside controlled airspace shall remain at least one nautical mile horizontally and 1,000 feet vertically away from cloud and in a flight visibility of at least five nautical miles :

Provided that at or below 3,000 feet above mean sea level or at or below 1,000 feet above the surface, whichever is the higher, this paragraph shall be deemed to be complied with if the aircraft is flown clear of cloud and in sight of the surface ".

1.17.4 The Fiji Air Limited Operations manual did not specify cloud ceiling and visibility minima for operations at Bua airstrip.

1.17.5 A little over one mile south of the accident site there is a peak at an elevation of 2,000 feet AMSL. A ridge extends to the west below this peak and the 1450 feet contour on the ridge is one nautical mile from the accident site on a bearing of 201°T. South of this ridge, on and to the west of this bearing of 201°T, the terrain is all below 1450 feet AMSL.

1.17.6 One submission was received in response to a public notice, in terms of Regulation 7 of the Civil Aviation (Investigation of Accidents) Regulations 1952, inviting any person who may desire to make written representations concerning the circumstances or causes of the accident.

1.17.7 The Fiji Aeronautical Communications Staff Association submitted that, " We are of the opinion that the disaster at Bua could possibly have been avoided if we had qualified (Aeronautical Communications) personnel manning Bua airport ". It was further stated that, " It is evident that the Department of Civil Aviation must . . . . . adopt a policy of manning most, if not all, of the domestic airports in Fiji by fully trained Flight Service Officers ".

2. ANALYSIS

2.1 The physical evidence at the accident site indicated that the aircraft was in level flight when the accident occurred. Having regard to the " in trim " airspeed determined during the flight test subsequently carried out (see paragraph 1.16.1) it is considered that at impact the aircraft was flying at an airspeed approximating its normal cruise speed of 125 knots.

2.2 In view of the quantity of fuel in the main tanks at departure, the selection of tip tanks during the latter stage of the flight would be normal. The positions of the tanks at the accident site and the impact damage which they had sustained indicated that they had contained fuel at the time of the accident.

2.3 The damage to the blades of both propellers supported the view that both engines had been operating under power at the time of impact.

2.4 When departing from Nausori the pilot had been requested to " report arrival Bua ". In normal circumstances there was no requirement for him to contact ATC during the flight and the lack of communication was not considered to be significant in relation to the accident.

2.5 Under calm conditions, at a true airspeed of 125 knots, the en route time from Nausori to Bua would be 34 minutes. If one minute was allowed to set heading after take-off from runway 10, the estimated time of arrival of DQ-FBO at Bua would have been 1021 hours. To have reached the accident site at 1017 hours the aircraft would need to have experienced an average tailwind component of approximately 22 knots over the entire flight. There was no evidence that a tailwind component of this strength existed over the entire route which suggests that the accident occurred at some time after the seventeen minutes past ten shown on the aircraft clock.

2.6 A review of the meteorological information and reports of actual conditions received after the accident, suggested that the flight would have been operating in light and variable wind conditions over much of the route. However it is probable that a west to south-westerly wind affected the flight in the vicinity of Vanua Levu.

2.7 To approach the accident site in level flight clear of the terrain immediately to the south (see paragraph 1.17.5) the aircraft must have been on a track of not less than 021°T i.e. at least 18° to the east of the direction of the Nausori-Bua track (003°T). The upper winds measured at Nadi and the pilot reports mentioned in paragraphs 1.7.7 and 1.7.11 support the view that a strong west to southwest wind probably drifted the aircraft to the east as it approached Bua. A westerly wind of 30 knots (on the beam) would have given 14° of starboard drift while a wind strength of 40 knots would have given 18° of starboard drift.

2.8 It was not practicable to determine the precise aircraft heading at impact or the extent to which the nose of the aircraft had yawed to the left after striking the trees. The subsequent trajectory of the fuselage would have been different from the direction in which the aircraft was tracking prior to the accident.

2.9 The spacing of weather reporting stations and the frequency of observations was such that the conditions over the route from Nausori to Bua could not have been forecast in any greater detail than was stated in the Fiji area forecast. The actual wind conditions experienced by flight PC71 in the vicinity of Vanua Levu, probably were significantly different from the forecast for variable winds at 10 knots. However, had the pilot maintained visual reference to the surface, he would have been able to observe the coastline of the island of Vanua Levu as the aircraft proceeded to the east of the desired track and thus been able to take appropriate action to keep clear of the terrain.

2.10 The actual weather conditions reported by the pilots of other aircraft suggest that flight PC71 would have experienced showery conditions while flying along the coast of Viti Levu to Tailevu Point. From Tailevu Point the flight had a distance of some 38 nautical miles to cover over water before approaching the south-west coast of Vanua Levu in the vicinity of Nabouwalu. While the pilot probably complied with the Visual Flight Rules when he first encountered reduced visibility conditions, it is apparent that in the latter stages he continued the flight with no visual reference to the surface and thus was unaware that the aircraft was over land.

2.11 The regulations concerning route training (see paragraph 1.17.2) did not require a specific number of flights over any particular route and the relevant requirements had been complied with so that the Operator was satisfied that Captain was competent to fly VFR in command on the Nausori-Bua-Nausori route. While taking into account Captain's flying experience it is nevertheless considered that the training given was not sufficient.

2.12 Captain was authorised as a pilot in command with Fiji Air Limited after 6 days of training and supervised route experience which he received over a period of 8 days. The completion of several circuits and landings at Bua would have familiarised him with the terrain in the area and with the approaches to the aerodrome in VFR conditions. However, one flight under supervision to Bua and 6 days of training on the Company's route network is not considered to have been sufficient to provide an adequate knowledge of the route or of the local meteorological conditions. Moreover, the pilot had received no training in the procedures for IFR operation on the Nausori-Bua-Nausori route. In relation to these particular points the training given was inadequate.

2.13 The section of the Visual Flight Rules applicable to the accident flight required only that the aircraft be flown clear of cloud and in sight of the surface. While it is possible to navigate visually under these conditions when following a well defined landmark such as a reef or a coastline, it is not considered that this minimum visibility criterion provides an acceptable level of safety for VFR public transport operations particularly over water, as the rules permit the pilot to operate with virtually no forward visibility in flight. In addition, no cloud ceiling and visibility minima had been specified for landing at Bua. Despite any instructions given by the Operator to Captain not to "push on into bad weather" (see paragraph 1.5.8) it is probable that this lack of specific operating criteria, both en route and for landing, was a factor in the pilot continuing the flight into deteriorating weather conditions.

2.14 The aircraft was equipped for IFR operation and, although the Operator stated that the Captain had been instructed not to fly in instrument flight conditions, this should not have influenced the pilot against proceeding IFR as a safety measure in the event of losing sight of the surface. There was no evidence at the accident site to suggest that the aircraft was in a climbing attitude or in a turn when it struck the trees.

2.15 It is possible that if a report of the current meteorological conditions at Bua had been available to him, the pilot may have decided not to operate flight PC71 or alternatively to have returned to Nausori. However, as there was no accurate record of the actual weather conditions existing at Bua at the relevant times, or of the actual weather conditions encountered by flight PC71 while en route, this must be a matter of conjecture.

3. CONCLUSIONS

3.1 The Operator held approval from the Fiji Air Transport Licensing Board to operate the air service.

3.2 The Operator had complied with the relevant requirements regarding aircraft equipment for the conduct of a scheduled public transport service.

3.3 Appropriate steps were taken in making enquiries when DQ-FBO became overdue and a full scale search and rescue operation was mounted when these enquiries failed to locate the aircraft.

3.4 The Millibar sub-scale of the pilot's altimeter was correctly set at the Nausori QNH of 1015 mbs.

3.5 The aircraft all up weight was within limits. The fact that the aircraft C of G was aft of the rear limit was not significant.

3.6 The aircraft certificate of airworthiness and certificate of maintenance were both valid.

3.7 The aircraft was airworthy up to the moment of impact.

3.8 Although the Operator was satisfied that the regulations concerning route training had been complied with, it is considered that Captain had not been given adequate training on the Nausori-Bua-Nausori route before he was authorised as a pilot in command.

3.9 The Visual Flight Rules did not specify a minimum visibility requirement applicable to the flight.

3.10 The Operator had not specified cloud ceiling and visibility minima for landing at Bua.

3.11 The short period of supervised en route experience which the pilot in command received together with the lack of specific guidance on operating criteria to be observed are considered to have been contributory factors in the accident.

3.12 The aerodrome facilities available were satisfactory for VFR operations to and from Bua aerodrome and for VFR and IFR operations to and from Nausori aerodrome.

3.13 While there was no communications facility at Bua aerodrome it cannot be concluded that, if this facility had been available, the accident would have been averted by reason of that fact alone.

3.14 While the weather in the vicinity of Bua at the time of the accident was apparently worse than the area forecast had indicated, this lack of accuracy of the forecast was not considered to have been a major factor in the accident because the pilot could have elected to return to Nausori either VFR or IFR.

3.15 The pilot in command failed to comply with the Visual Flight Rules and continued the flight without visual reference to the surface.

3.16 Probable cause : The accident was caused by the pilot in command continuing the flight into conditions where he lost visual reference with the surface which resulted in the aircraft colliding with the terrain.

OBSERVATIONS

The circumstances of this accident have been investigated with the object of avoiding similar accidents in the future.

The accident flight was being conducted under the Visual Flight Rules and these observations and recommendations are concerned with maintaining adequate operational safety standards under these Rules.

There are no aspects of airworthiness standards which require consideration as no deficiencies were apparent in relation to aircraft maintenance and airworthiness.

The concept of using an IFR equipped aircraft flown by an instrument rated pilot to operate under the Visual Flight Rules to domestic aerodromes on outlying islands is basically sound. The minimum fuel carried is sufficient to enable the aircraft to return to the aerodrome of departure with normal reserves, including fuel to an alternate aerodrome if necessary, so that under adverse weather conditions the pilot has the option of returning either VFR or IFR, to his departure point or of proceeding to an alternate aerodrome.

Three airports in the Fiji Group are equipped for IFR operation and the level of safety attainable under these conditions is related to the criteria stipulated in the Visual Flight Rules, the aerodrome meteorological minima specified and the accuracy of the terminal weather forecasting for the IFR equipped airports. However the ultimate responsibility for the safety of a flight rests with the pilot in command and the decisions which he makes when planning and operating the flight.

Any improvements which may be made to provide the pilot in command with more accurate or up to date information would be desirable but it is considered that compliance with operational standards and adequate surveillance of these standards are prerequisites in maintaining safety in airline operations.

SAFETY RECOMMENDATIONS

It is recommended that :

1. The Visual Flight Rules be amended to specify a minimum visibility requirement for aircraft operating outside controlled airspace at or below 3000 feet above mean sea level or at or below 1000 feet above the surface, whichever is the higher.
2. For public transport operations under the Visual Flight Rules, cloud ceiling and visibility minima be specified for take-off and landing at each aerodrome or airstrip ; such minima to be appropriate to the flight characteristics of the aircraft type to be operated.
3. A minimum en route altitude be stipulated for VFR flight.
4. Suitable minimum requirements be specified for the training of captains to ensure that they are thoroughly familiar with en route procedures and operations of the aircraft before being cleared as pilots in command. The pilot should have obtained the necessary Fiji flight crew licence and ratings during the training period.
5. In view of the expansion of Fiji airline services, the Director of Civil Aviation review the adequacy of the operational supervision exercised by his Department over airlines conducting public transport flights under Fiji regulatory requirements.
6. Weather reporting stations be on or adjacent to domestic aerodromes. In addition a recent actual weather report from a destination aerodrome be available to the pilot in command before departure, particularly on the first flight of the day.
7. A trim sheet or other suitable quick reference data be used to ascertain the aircraft centre of gravity when completing the loadsheet.
8. Inertia actuated Emergency Locator Transmitters be carried as standard equipment on public transport aircraft.

ICAO Ref.: 227/79.

No. 7

McDonnell-Douglas DC-10-10, N110AA, accident at Chicago-O'Hare International Airport, Illinois, United States, on 25 May 1979.
Report No. NTSB-AAR-79-17 dated 21 December 1979
released by the National Transportation Safety Board, United States.

SYNOPSIS

About 1504 c.d.t., May 25, 1979, American Airlines, Inc., Flight 191, a McDonnell-Douglas DC-10-10 aircraft, crashed into an open field just short of a trailer park about 4,600 ft northwest of the departure end of runway 32R at Chicago-O'Hare International Airport, Illinois.

Flight 191 was taking off from runway 32R. The weather was clear and the visibility was 15 miles. During the takeoff rotation, the left engine and pylon assembly and about 3 ft of the leading edge of the left wing separated from the aircraft and fell to the runway. Flight 191 continued to climb to about 325 ft above the ground and then began to roll to the left. The aircraft continued to roll to the left until the wings were past the vertical position, and during the roll, the aircraft's nose pitched down below the horizon.

Flight 191 crashed into the open field and the wreckage scattered into an adjacent trailer park. The aircraft was destroyed in the crash and subsequent fire. Two hundred and seventy-one persons on board Flight 191 were killed; two persons on the ground were killed, and two others were injured. An old aircraft hangar, several automobiles, and a mobile home were destroyed.

The National Transportation Safety Board determines that the probable cause of this accident was the asymmetrical stall and the ensuing roll of the aircraft because of the uncommanded retraction of the left wing outboard leading edge slats and the loss of stall warning and slat disagreement indication systems resulting from maintenance-induced damage leading to the separation of the No. 1 engine and pylon assembly at a critical point during takeoff. The separation resulted from damage by improper maintenance procedures which led to failure of the pylon structure.

Contributing to the cause of the accident were the vulnerability of the design of the pylon attach points to maintenance damage; the vulnerability of the design of the leading edge slat system to the damage which produced asymmetry; deficiencies in Federal Aviation Administration surveillance and reporting systems which failed to detect and prevent the use of improper maintenance procedures; deficiencies in the practices and communications among the operators, the manufacturer, and the FAA which failed to determine and disseminate the particulars regarding previous maintenance damage incidents; and the intolerance of prescribed operational procedures to this unique emergency.

1. FACTUAL INFORMATION

1.1 History of the Flight

At 1459 c.d.t.,[1/] May 25, 1979, American Airlines, Inc., Flight 191, a McDonnell-Douglas DC-10 series 10 aircraft (DC-10-10) (N110AA), taxied from the gate at Chicago-O'Hare International Airport, Illinois. Flight 191, a regularly scheduled passenger flight, was en route to Los Angeles, California, with 258 passengers and 13 crewmembers on board. Maintenance personnel who monitored the flight's engine start, push-back, and start of taxi did not observe anything out of the ordinary.

The weather at the time of departure was clear, and the reported surface wind was 020° at 22 kns. Flight 191 was cleared to taxi to runway 32 right (32R) for takeoff. The company's Takeoff Data Card showed that the stabilizer trim setting was 5° aircraft noseup, the takeoff flap setting was 10°, and the takeoff gross weight was 379,000 lbs. The target low-pressure compressor (N_1) rpm setting was 99.4 percent, critical engine failure speed (V_1) was 139 kns indicated airspeed (KIAS), rotation speed (V_R) was 145 KIAS, and takeoff safety speed (V_2) was 153 KIAS.

Flight 191 was cleared to taxi into position on runway 32R and hold. At 1502:38, the flight was cleared for takeoff, and at 1502:46 the captain acknowledged, "American one ninety-one under way." Company personnel familiar with the flightcrew's voices identified the captain as the person making this call and the ensuing V_1 and V_R speed callouts on the cockpit voice recorder (CVR).

The takeoff roll was normal until just before rotation at which time sections of the left, or No. 1, engine pylon structure came off the aircraft. Witnesses saw white smoke or vapor coming from the vicinity of the No. 1 engine pylon. During rotation the entire No. 1 engine and pylon separated from the aircraft, went over the top of the wing, and fell to the runway.

Flight 191 lifted off about 6,000 ft down runway 32R, climbed out in a wings-level attitude, and reached an altitude of about 300 ft above the ground (a.g.l.) with its wings still level. Shortly thereafter, the aircraft began to turn and roll to the left, the nose pitched down, and the aircraft began to descend. As it descended, it continued to roll left until the wings were past the vertical position.

Flight 191 crashed in an open field and trailer park about 4,600 ft northwest of the departure end of runway 32R. The aircraft was demolished during the impact, explosion, and ground fire. Two hundred and seventy-one persons on board Flight 191 were killed, two persons on the ground were killed, and two persons on the ground sustained second- and third-degree burns.

The aircraft crashed about 1504, during daylight hours; the coordinates of the crash site were 42°00'35"N, 87°55'45"W.

1/ All times herein are central daylight time, based on the 24-hour clock.

1.2 Injuries to Persons

Injuries	Crew	Passengers	Others
Fatal	13	258	2
Serious	0	0	2
Minor/None	0	0	0

1.3 Damage to Aircraft

The aircraft was destroyed.

1.4 Other Damage

An old aircraft hangar, several automobiles, and a mobile home were destroyed.

1.5 Personnel Information

All flight and cabin personnel were qualified.

1.6 Aircraft Information

Flight 191, a McDonnell-Douglas DC-10-10, N110AA, was owned and operated by American Airlines, Inc., and was powered by three General Electric CF6-6D engines. According to the manufacturer, the left engine weighed 11,612 lbs, the pylon, 1,865 lbs, for a total engine-pylon assembly weight of 13,477 lbs. With the loss of the engine pylon structure, the aircraft's center of gravity (c.g.) moved aft 2 percent to about 22 percent mean aerodynamic chord (MAC). The resultant c.g. was within the forward (16.4 percent MAC) and aft (30.8 percent MAC) c.g. limits. The lateral c.g. shift was 11.9 inches to the right.

1.7 Meteorological Information

At the time of the accident, the weather at the airport was clear. The surface observations at O'Hare International were as follows:

1451, surface aviation: Clear, visibility—15 mi, weather--none, temperature--63°F, dewpoint--29°F, winds--020° at 22 kns, altimeter--30.00 inHg.

1511, local: Clear, visibility--15 mi, weather--none, temperature--63° F, dewpoint--29° F, winds--020° at 19 kns gusting to 28 kns, altimeter--30.00 inHg., remarks--aircraft mishap.

1.8 Aids to Navigation

Not applicable.

1.9 Communications

There were no known communications malfunctions.

1.10 Aerodrome Information

Chicago-O'Hare International Airport is located 16 mi northwest of downtown Chicago, Illinois, and is served by seven runways. Runway 32R is 10,003 ft long and 150 ft wide, and has a concrete surface. The runway elevation is 649 ft mean sea level (m.s.l.) at its southeast end and 652 ft m.s.l. at its northwest end.

1.11 Flight Recorders

The aircraft was equipped with a Fairchild Model A-100 CVR, serial No. 2935. The CVR was recovered and brought to the Safety Board's laboratory where a transcript of the recording was prepared. The recording was incomplete because of the loss of electrical power to the recorder during aircraft rotation. However, the aircraft's gross weight, stabilizer trim setting, V_1, and V_R callouts were recorded.

The aircraft was equipped with a Sundstrand digital flight data recorder (DFDR), serial no. 2298. The recorder had been damaged structurally, but there was no fire or heat damage. The recording tape was broken; upon removal from the recorder the tape was spliced together and a readout was made. Two 6-sec areas of data were damaged because of the breaks in the tapes; however, most of these data was recovered.

The DFDR recorded 50 sec of data during the takeoff roll and 31 sec of airborne data before the recording ended. (See appendix H.) The DFDR readout showed that the stabilizer trim setting for takeoff was 6.5° aircraft noseup. The DFDR's tolerance for this parameter is ± 1°. Because of unusual aircraft attitudes during the last few seconds of the flight, the recorded altitude and airspeed data were not correct. Therefore, the DFDR altitude and indicated airspeed values cited hereafter have been corrected for the position errors resulting from the aircraft's attitudes during the last few seconds of the descending flight.

Correlation of the DFDR and CVR recordings disclosed that the flightcrew had set the flaps and stabilizer trim at 10° and about 5° aircraft noseup, respectively, for takeoff. A rolling takeoff was made, takeoff thrust was stabilized at 80 KIAS, and left rudder and right aileron were used to compensate for the right crosswind. The V_1 and V_R callouts were made about 2 sec after these speeds were recorded by the DFDR. The elevator began to deflect up at V_R. The aircraft began to rotate upward immediately and continued upward at a rate of 1.5° per sec. Flight 191 accelerated through V_2 speed during rotation and before it lifted off the runway. The last stable takeoff thrust on the No. 1 engine was recorded 2 sec before liftoff. One second later, the word "damn" was recorded on the CVR, and then the CVR ceased operating.

One second before liftoff and simultaneous with the loss of the CVR and the No. 1 engine's parameters, the DFDR ceased recording the positions of the left inboard aileron, left inboard elevator, lower rudder, and Nos. 2 and 4 left wing leading edge slats. The DFDR continued to record all other parameters

including the position of the upper rudder, the outboard aileron, the outboard elevator, and the No. 4 leading edge slats on the right side of the aircraft. The electrical power for the CVR and the sensors for the lost DFDR functions were all derived from the aircraft's No. 1 a.c. generator bus.

Flight 191 became airborne about 6,000 ft from the start of the takeoff roll and remained airborne for 31 sec. It lifted off at V_2 + 6 KIAS and at 10° pitch attitude. Two seconds after liftoff, the DFDR reading for the No. 1 engine's N_1 was zero, the No. 2 engine's N_1 speed was increasing through 101 percent, and the No. 3 engine's N_1 was essentially at the takeoff setting.

The flight lifted off in a slight left wing-down attitude. Application of right wing-down aileron and right rudder restored the flight to a wings-level attitude and the heading was stabilized between 325° and 327°. The flight maintained a steady climb about 1,150 feet per minute (fpm) at a 14° noseup pitch attitude--the target pitch attitude displayed by the flight director for a two-engine climb. During the climb, the No. 2 engine N_1 speed increased gradually from 101 percent to a final value of 107 percent; the No. 3 engine N_1 speed did not change appreciably from the takeoff setting. During the initial part of the climb, the aircraft accelerated to a maximum speed of 172 KIAS; it reached this value about 9 sec after liftoff and about 140 ft a.g.l.

Flight 191 continued to climb about 1,100 fpm. The pitch attitude and heading were relatively stable. Right wing-down aileron and right rudder were used to control and maintain the heading and the roll attitude during the climb in the gusty right crosswind.

During the climb, the aircraft began to decelerate from 172 KIAS at an average rate of about 1 kn per second. At 20 sec after liftoff, at 325 ft a.g.l. and 159 KIAS, the flight began to roll to the left and passed through 5° left wing down. The left roll was accompanied by increasing right-wing-down aileron deflection. At this point, the previously stabilized right rudder deflected suddenly to zero, remained at zero for 1 sec, and then moved toward its previous deflection. The flight began to turn to the left, and the left roll increased even though increasing right rudder and right-wing-down aileron deflections were being applied. At 325 ft a.g.l. the flight had turned through the runway heading and was rolling to the left at 4° per second. The right rudder deflection increased during the turn. The previously stable pitch attitude began to decrease from 14° even though the elevator was being increased to the full aircraft noseup deflection. The maximum pitch rate of about 12° per second was reached just before the crash.

Flight 191 continued to roll and turn to the left despite increasing right rudder and right-wing-down aileron deflections. Three seconds before the end of the DFDR tape, the aircraft was in a 90° left bank and at a 0° pitch attitude. The DFDR recording ended with the aircraft in a 112° left roll and a 21° nosedown pitch attitude with full counter aileron and rudder controls and nearly full up elevator being applied.

DFDR longitudinal and vertical acceleration data were integrated to determine the headwind components at points where the aircraft attained certain speeds and where it lifted off; to establish an altitude profile; and to determine the location where the DFDR stopped. These data showed that the DFDR ceased operation 14,370 ft from the southeast end of runway 32R and 820 ft left of the runway's extended centerline. Examination of the crash site showed that the first point of impact was 14,450 ft beyond the southeast end of runway 32R and 1,100 ft left of its extended centerline. Based on these data and the corrected altitudes, the DFDR ceased operating at impact. The flight reached a maximum altitude of 350 ft a.g.l.

1.12 Wreckage and Impact Information

Flight 191 struck the ground in a left wing-down and nosedown attitude. The left wingtip hit first, and the aircraft exploded, broke apart, and was scattered into an open field and a trailer park. The disintegration of the aircraft structure was so extensive that little useful data were obtained from postimpact examination of the wreckage with the exception of the No. 1 pylon, which was found off the right side of runway 32R. (See figure 1.)

Investigators located and documented identifiable aircraft components. Except for the No. 1 engine and pylon, portions of the engine cowling, and a part of the leading edge of the wing directly above the pylon, the aircraft wreckage came to rest in the open field and trailer park. (See appendix D.)

The first marks made by engine contact of the No. 1 engine and pylon with the runway began about 19 ft to the right of the centerline lights and about 6,953 ft beyond the southeast end of runway 32R. Other parts of engine and pylon structure were located in this area; however, no spoiler actuators or hydraulic lines were found.

The pylon is attached to the wing using spherical ball joints in three different structural elements. Two of the spherical joints are aligned vertically in a forward bulkhead which is attached to structure in the wing forward of the front spar. Another spherical joint behind the forward bulkhead transmits thrust loads from pylon structure into a thrust link which in turn is connected through another spherical joint to structure on the lower surface of the wing. The third attachment point is a spherical joint in the pylon aft bulkhead which attaches to a clevis mounted on the underside of the wing. The pylon forward bulkhead and portions of the flange from the pylon aft bulkhead either remained with the separated No. 1 pylon or were scattered along the runway. (See figures 2 and 3.) The No. 1 pylon's aft clevis attach assembly and portions of the pylon aft bulkhead, wing thrust angle assembly and thrust link, and pylon forward bulkhead attach assembly remained with the wing.

The pylon forward bulkhead was bent forward about 30° and most of the bolts which held the bulkhead upper plates were missing. The upper 12 inches of the forward plate were bent forward an additional 10° to 15°. The aft plate was broken below the thrust fitting connection, and a large piece of the upper left corner was missing.

Figure 1. Accident site.

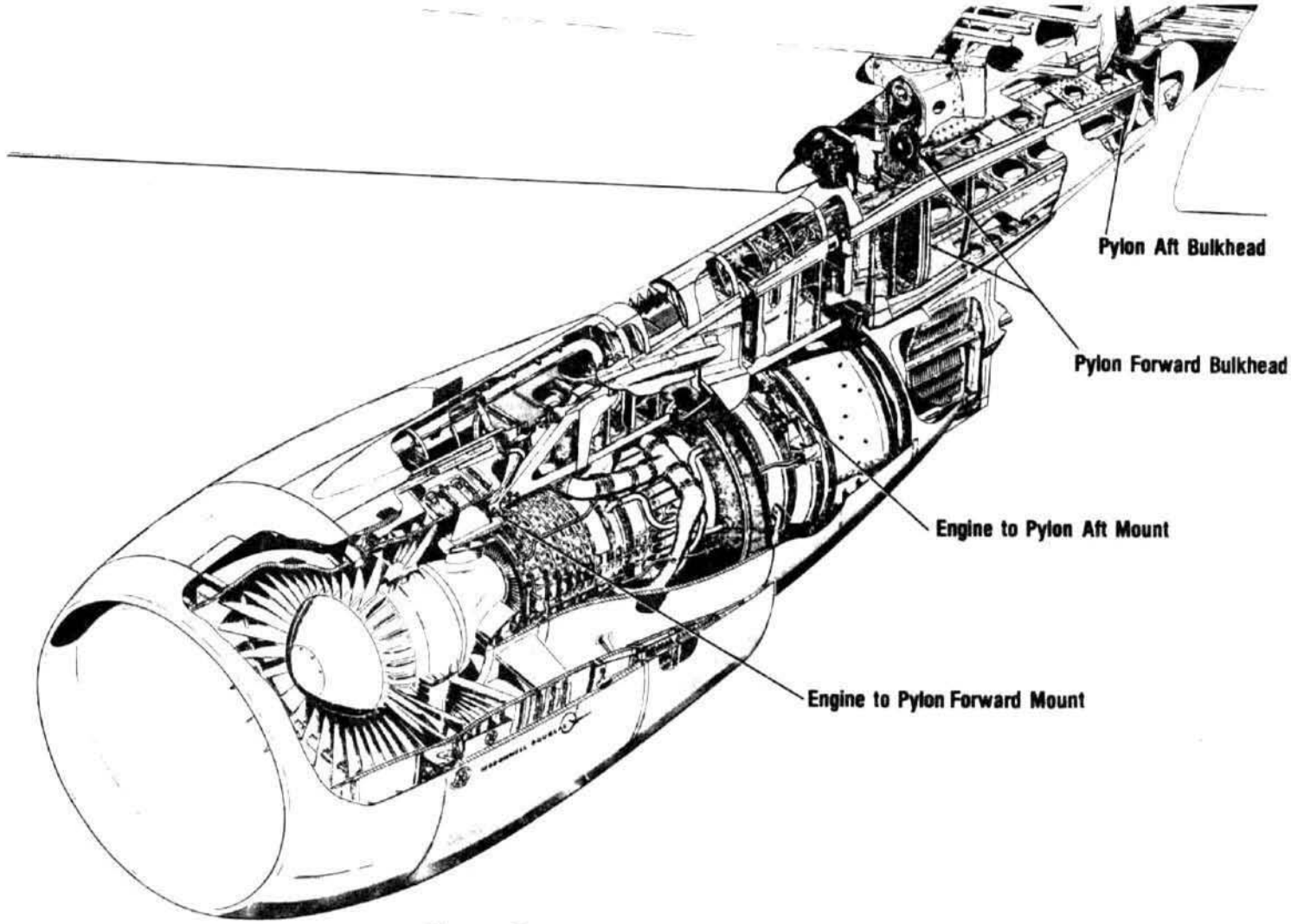


Figure 2. Engine and pylon assembly.

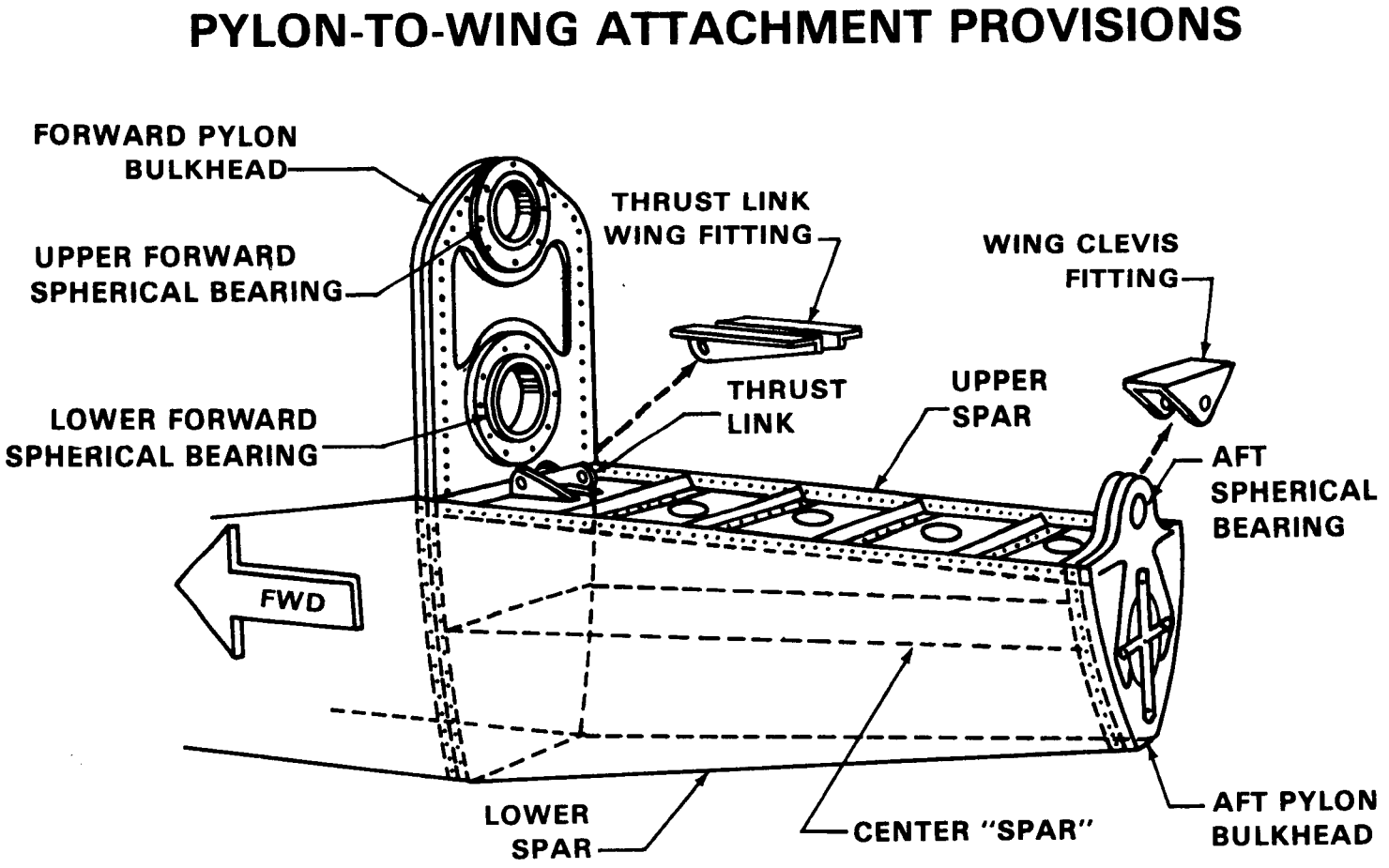


Figure 3. Pylon assembly.

The wing's forward support fitting, which attached the pylon forward bulkhead to the wing at the upper and lower plugs and spherical bearings, was found at the main wreckage site. The upper and lower plugs and their attaching hardware were intact, and the upper and lower spherical bearings were attached to the fitting.

The pylon thrust fitting remained attached to the forward portion of the pylon's aft upper spar web. The pylon thrust link, which attached the pylon thrust fitting to the wing thrust angles, was found at the main wreckage site attached to a portion of the wing thrust angles. Its forward spherical bearing was cocked to the extreme left, and a segment of the bearing which had broken away was found on the runway.

The thrust bushing bolt had broken in two parts, both of which were found in the grass adjacent to the runway. The bolt nut was attached to one of the broken pieces, and the faces of the nut were gouged severely. Except for one lubrication retainer washer, which was not found, the remaining portions of the thrust bushing bolt assembly were found along the runway. One shim spacer from the assembly was crushed severely while the other was relatively undamaged.

The upper two-thirds of the pylon aft bulkhead separated from the flanges around its periphery and was found in the wreckage. The top two pieces of its attach lugs had separated from the bulkhead, and the aft side of the bulkhead was gouged heavily near the lower edge of the wing clevis lug, which attached the aft bulkhead to the wing. The wing clevis was attached to the wing. The aft bulkhead's spherical bearing was attached to the clevis, and the separated pieces of the aft bulkhead's attach lugs were found on top of the spherical bearing.

The Nos. 2 and 3 engines were located in the main wreckage. The damage to the engines indicated that they were operating at high rpm at impact. All three engines were taken to the American Airline's Maintenance Facility at Tulsa, Oklahoma, where they were torn down and examined. There was no evidence of any preimpact malfunctions.

The examinations of the main and nose landing gears and actuators indicated that the gear was down and locked at impact. The left and right stabilizer jackscrews were recovered and the distance between the upper surfaces of the jackscrews' drive nuts and the lower surfaces of the actuators' upper stops was measured. These measurements indicated that the stabilizer was positioned at 5.71° aircraft noseup.

Examination of the hydraulic system components did not reveal any evidence of internal operating distress. The control valve of the 2-1 nonreversible motor pump was in the open position, indicating that the No. 2 hydraulic system was driving the No. 1 hydraulic system's pump.

All eight flap actuators were recovered, and investigators attempted to verify the position of the trailing edge flaps by measuring the extension of the flap actuator pistons. The piston extensions were compared to those of another aircraft with flaps extended to 10°. Based on this comparison, some degree of

flap extension was probable, but the actual position could not be established. However, the DFDR data showed that the flaps were set at 10°.

A 3-ft section of the left wing's leading edge, just forward of the point where the forward part of the pylon joined the wing, was torn away when the engine pylon assembly separated from the aircraft. The No. 1 and No. 3 hydraulic system's extension and retraction lines and the followup cables for the left wing's outboard slat drive actuators were severed. Thirty-five of the 36 leading edge slat tracks were examined. The examination disclosed that at impact the left wing's outboard slats were retracted, while the left wing's inboard slats and the right wing's inboard and outboard slats were extended to the takeoff position.

The examination of the cockpit instruments did not disclose any usable information.

1.13 Medical and Pathological Information

A review of the autopsies and toxicological examinations of the flightcrew disclosed no evidence of preexisting physiological problems which could have affected their performance.

1.14 Fire

The aircraft was subjected to severe ground fire.

1.15 Survival Aspects

This accident was not survivable because impact forces exceeded human tolerances.

1.16 Tests and Research

1.16.1 Study of Photographs

Five photographs taken of Flight 191's departure by two cameras--one in the terminal and one onboard a DC-10 on final approach to runway 9R--were sent to Lockheed's Palo Alto Research Laboratories for a Photo-Image Enhancement Study to determine the position of the flight controls. The process produced black and white images containing expanded variations of gray shading which, in the absence of the enhancement process, would be too subtle for the eye to distinguish. Based on the study of these photo-images, the following observations were made: (1) The tail assembly was not damaged; (2) the nose gear was down during the initial climbout and before the onset of roll; (3) spoilers Nos. 1, 3, and 5 were extended on the right wing; and (4) the trailing edge of the right wing inboard aileron was up. Although the position of the slats was difficult to determine, the left wing inboard slats appeared to be extended, and the position of all other control surfaces appeared to be the same as recorded by the DFDR. The pitch and roll attitudes of the aircraft were extrapolated from the photographs, and extrapolations agreed closely with those recorded by the DFDR.

1.16.2 Metallurgical Examinations and Postaccident Inspections of the DC-10 Fleet

N110AA's pylon aft bulkhead was examined at the Safety Board's metallurgical laboratory. The examination disclosed a fracture of the upper forward flange. (See figure 4.) The larger part of this fracture was just forward of the radius between the flange and forward bulkhead plane and was about 10 inches long in the inboard-outboard direction. (See figure 5.) The fracture characteristics were typical of an overload separation. Chevron and tear marks on the fracture indicated that the rupture progressed downward at the center of the flange, then in inboard and outboard directions on the flange. The bottom portion of the fracture exhibited smearing consistent with the compression portion of a bending fracture. The smear was more prevalent—about 6 inches long—in the thinner center portion of the upper flange structure, but became less prevalent at the outer ends of the fracture. The 10-inch-long fracture resulted from overstress. The overstress was initiated by the application of a downward bending moment at the center section of the flange just forward of the fracture plane. The surface of the fracture appeared to be relatively free of oxidation and dirt.

Fatigue cracking was evident at both ends of the fracture. At the inboard end, the fatigue progressed inboard and aft; then, it progressed downward and inboard to the upper inboard fastener that attached the forward section of the bulkhead to the aft section. The fatigue progressed past the fastener a short distance before exhibiting rapid overstress characteristics in the downward direction as it proceeded along the inboard side of the side flange radius of the forward flange section. At the outboard end of the fracture, the fatigue propagated forward and slightly outboard toward the most forward outboard hole in the upper flange. The total length of the overstress fracture and fatigue cracks was about 13 inches. The remainder of the fractures on the bulkhead and within the pylon structure resulted from overload.

The examination also disclosed that three shims were installed on the upper surface of the forward upper flange. Two shims (Part No. AUB-7034-25) were installed, one on the inboard top shoulder of the upper flange and one on the outboard top shoulder. These shims are about 2 inches long, 1 inch wide, and .063 inch thick. A 10-inch-long, .050-inch-thick shim was installed during production to fill a gap between the upper flange and upper spar web. (See figure 6.) The manufacturer's drawings specify that the AUB-7034-25 shim may be required along the side of the bulkhead; however, they do not indicate that shims may be required on the upper surface of the flange. The fatigue propagation on the inboard and outboard ends of the overstress fracture began in the area underneath the .063-inch-thick shims.

The aft fracture surface of the upper flange contained a crescent-shaped deformation which matched the shape of the lower end of the wing clevis. This deformation was in line with the vertical centerline of the aft bulkhead attachment hole as indicated by arrow "d" in figure 7. A deformation was noted in the lower surface of the aft wing support fitting's forward clevis lug in the area indicated by the brackets in figure 9. A small shallow gouge was apparent in the area of the arrow in figure 10. This gouge was in a position which would conform to a fastener location on the top flange assembly of the aft bulkhead.

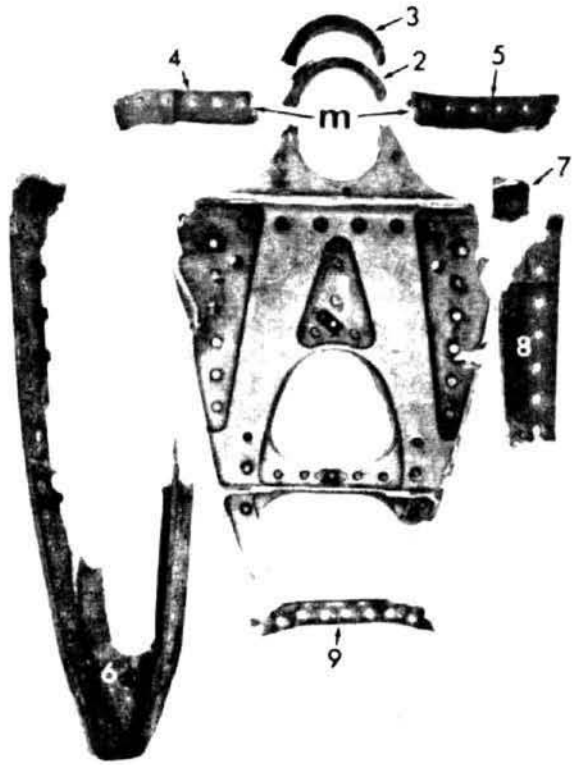


Figure 4. Overall view of the wing pylon aft bulkhead installation portions. Pieces are numbered for identification purposes and placed in relative locations as if intact.

Item 1.	Aft bulkhead center section piece view looking aft.
Items 2 & 3.	Upper lug ears.
Items 4 & 5.	Two pieces of the forward portion of the upper flange. Those pieces mated together along the fracture indicated by arrows "m."
Item 6.	Side flanges and lower portion of the aft bulkhead.
Item 7.	Piece of the flange at the upper outboard corner.
Item 8.	Portion of the outboard side flange.
Item 9.	Piece of the intermediate flange.

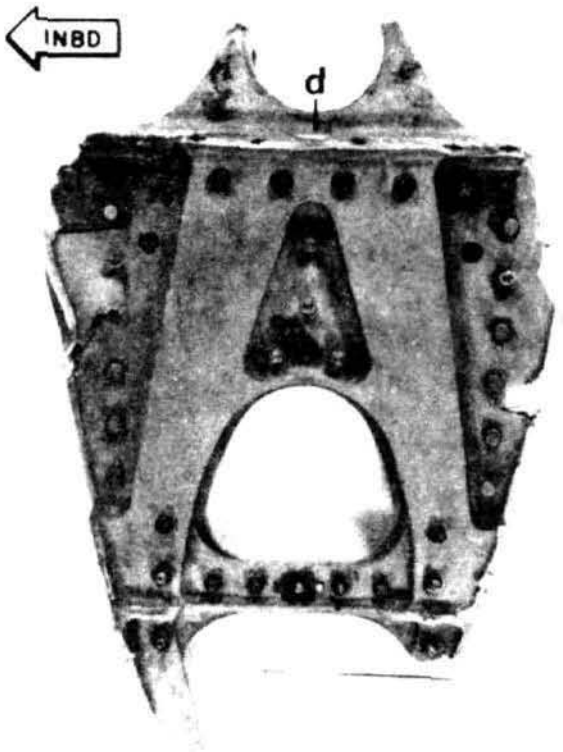


Figure 5. View of the aft bulkhead piece indicated as Item 1 in Figure 4.

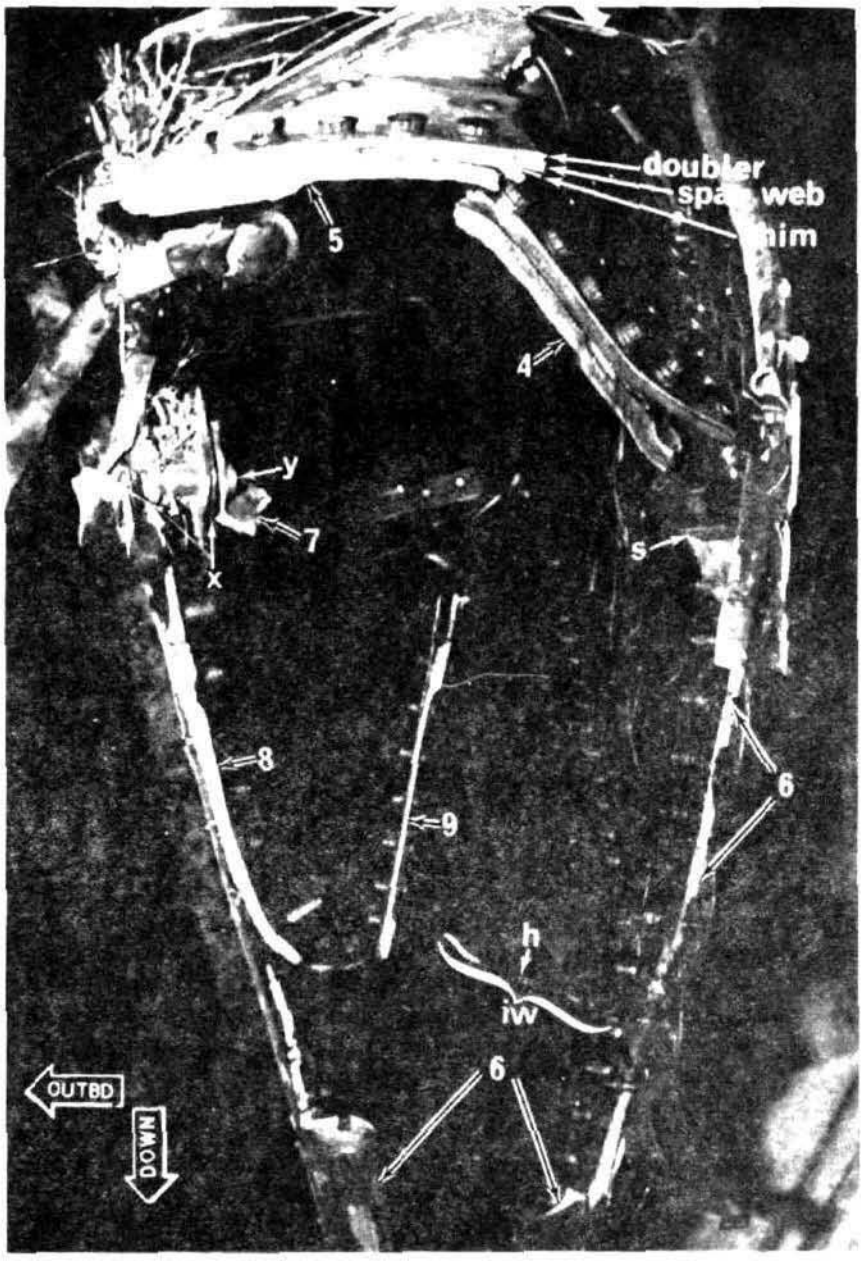


Figure 6. View looking forward and slightly down on the No. 1 pylon before disassembly of the flange pieces indicated as items 4 through 9 in Figure 4. Arrow "x" shows the shim on the outboard side between the angle and the top surface of the upper flange piece No. 7 held in place by the fastener arrowed "y". Arrow "s" shows the location for the shim on the inboard side between the angle and upper surface of the upper bulkhead flange.

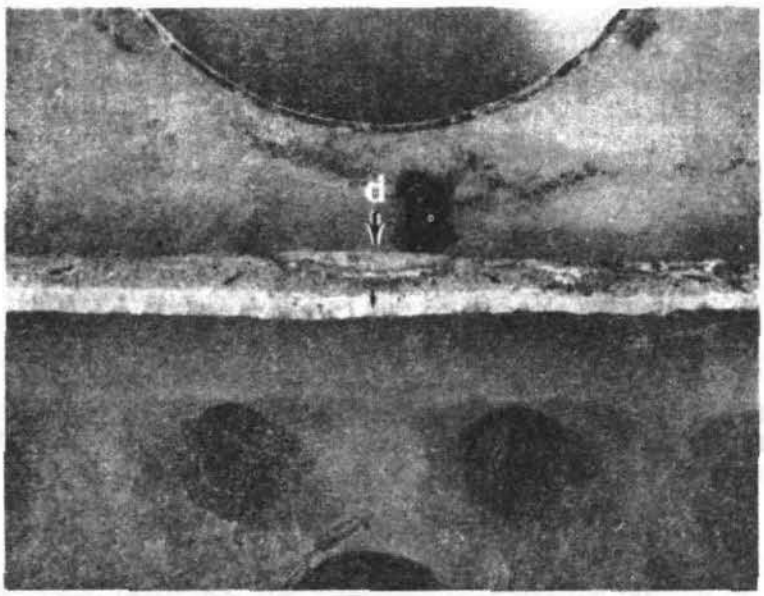


Figure 7. Closer view of fracture on upper flange in the area of deformation (arrow "d", see Figure 5).

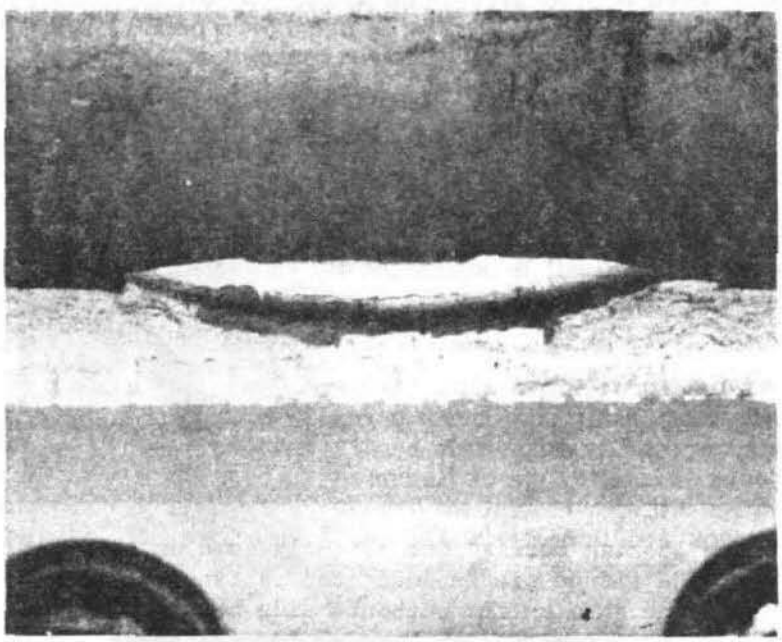


Figure 8. Detail of deformation denoted by arrows "d" in Figure 5 and 7.

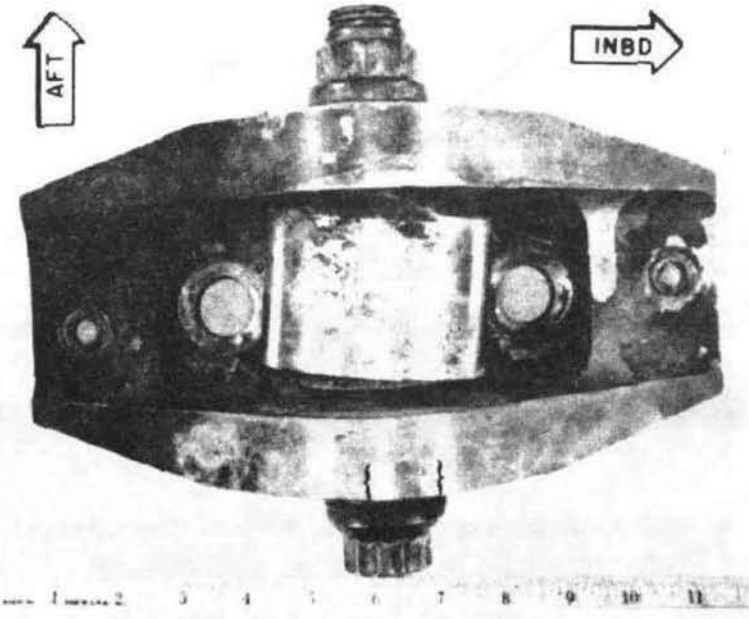


Figure 9. Overall view looking up on the wing mounted aft support fitting with spherical bearing attached.

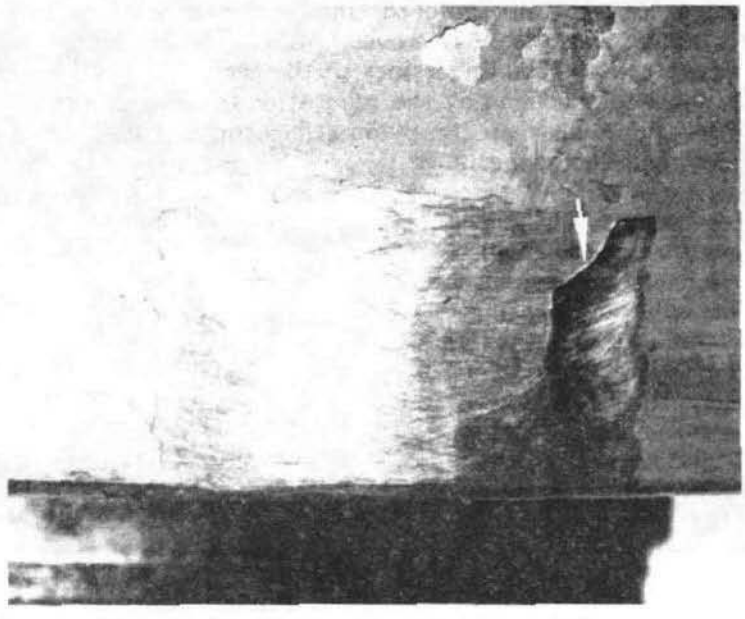


Figure 10. Close up view of damage in the area between the brackets of Figure 9.

The gouge appeared to be produced by a fastener head, hitting the clevis with a sliding movement. The upper flange aft fracture surface and radius appeared to have been deformed by the wing clevis' striking these surfaces in the downward direction.

The clearances between the upper flange surface and the bottom surface of the wing clevis were examined using the aft wing support fitting from N110AA and the aft bulkheads of another DC-10-10 (N119AA). (See figures 11 and 12.) With the aft spherical bearing and bushing in place, the vertical distance from the bottom of the clevis to the surface of the flange is about 0.5 inch. (See figure 11.) When the bushing was removed and the aft bulkhead moved up against the far inside portion of the wing fitting, the flange was displaced about 0.6 inch above its previous position. (See figure 12.) In this position, the lower portion of the clevis was about 0.1 inch below the fracture on N119AA's bulkhead. The 0.1 inch between the upper flange fracture's upper surface and the lower portion of the wing fitting clevis was the same as the vertical depth of the deformation found on N110AA's aft bulkhead.

Taking into account the stackup on the forward flange created by the spar web, doubler, and fasteners, the clearance between the bottom of the clevis and the top of the web fasteners could be about .005 to .045 inch. The addition of a shim would narrow the clearance, and taking into account all tolerances in the spherical bearing assembly, there could be an interference. A postaccident survey of the DC-10 fleet revealed seven pylons with such interference. McDonnell-Douglas had not established a standard minimum clearance between the bottom of the clevis and the top of the fastener.

Despite numerous searches of the runway and adjacent areas after the accident, investigators were not able to find one of the forward thrust bushing attachment's retainer washers. However, measurements between the mating portions on the fracture and the undersides of the thrust bolt head and nut as well as the physical evidence produced by the separation of the parts indicated that the missing washer was in place when the pylon separated and that the thrust bushing assembly had been installed properly.

After the accident, the Federal Aviation Administration (FAA) required a fleetwide inspection of the DC-10. During these inspections, discrepancies were found in the pylon assemblies. Among these discrepancies were variances in the clearances on the spherical bearing's fore and aft faces; variances in the clearance between the bottom of the aft wing clevis and the fasteners on the upper spar web; interferences between the bottom of the aft clevis and the upper spar web fasteners; pylons with either loose, failed, or missing spar web fasteners; and aft pylon bulkheads with upper flange fractures. The fractured flanges were found only on the DC-10-10 series aircraft.

During postaccident inspections, six DC-10's were found to have fractured upper flanges on the pylon aft bulkheads: Four American Airlines DC-10's--N106AA, N107AA, N118AA, N119AA-- and two Continental Airlines DC-10's--N68050, N68047.

The failure modes on the Continental Airlines' aircraft that were examined by metallurgists were similar to those found on the American Airlines' DC-10's. Of the two Continental fractures discovered during the postaccident

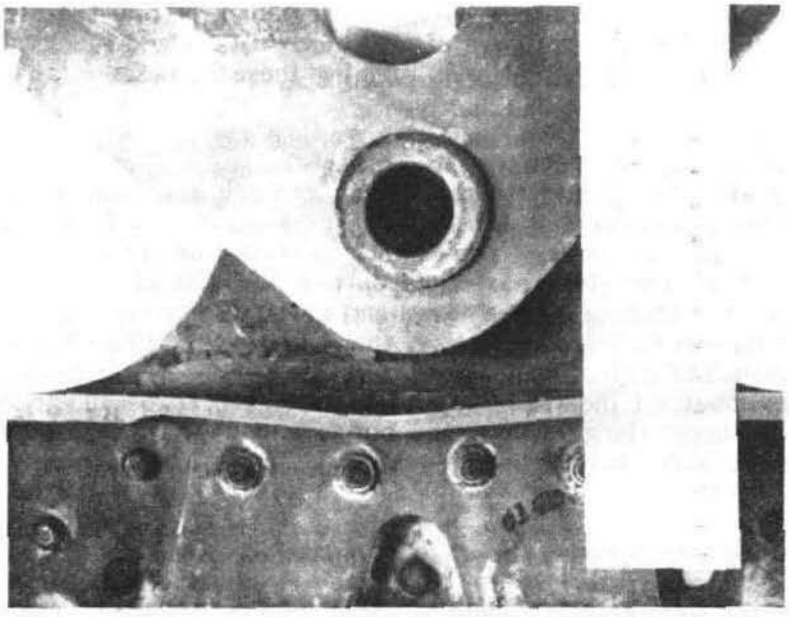


Figure 11. Wing pylon aft bulkhead from N119AA assembled to wing mounted aft support fitting of N110AA showing normal position of wing fitting clevis with respect to the upper forward flange of the aft bulkhead.
Note: The wing fitting on this figure is canted relative to the bulkhead to simulate the dihedral of the left wing relative to the bulkhead.

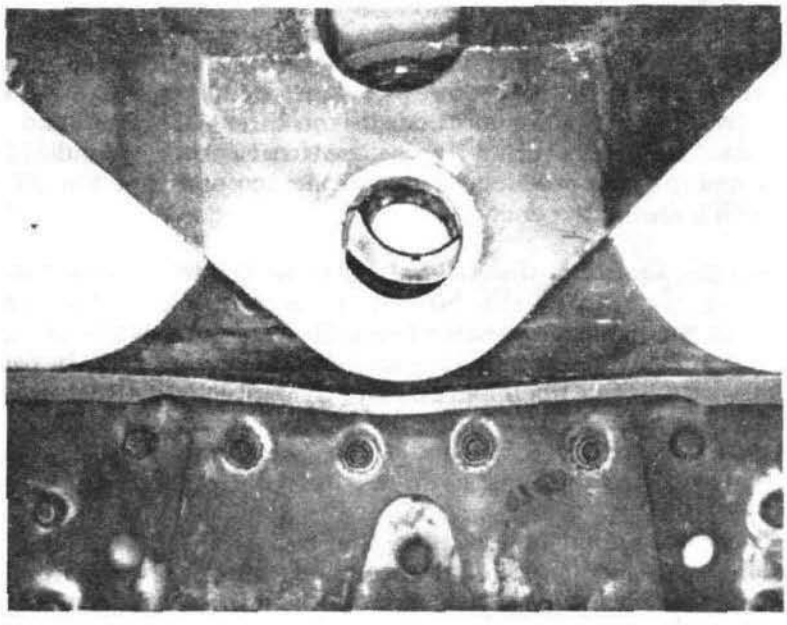


Figure 12. Same as Figure 11 except the attachment bushing was removed and the aft bulkhead was moved up against the far inside portion of the wing fitting. Note the location of the bottom portion of the wing fitting clevis with respect to the fracture on the bulkhead.

inspections, one crack was 6 inches long, and the other 3 inches long; neither crack showed any evidence of fatigue propagation.

The investigation also disclosed that two other Continental Airlines DC-10's--N68041, N68049--had had fractures on their upper flanges. These two aircraft were damaged on December 19, 1978, and February 22, 1979, respectively. The damage was repaired and both aircraft were returned to service. In addition, a United Airlines' DC-10, N1827U, was discovered to have a cracked upper spar web on its No. 3 pylon and 26 damaged fasteners.

The damaged pylon aft bulkheads of the four other American Airlines' DC-10's were also examined at the Safety Board's metallurgical laboratory. Each of these aft bulkheads contained visible cracks and obvious downward deformations along their upper flanges. The shortest crack appeared to be on the N107AA bulkhead and the longest crack—about 6 inches—was on the N119AA bulkhead. The crack on the N119AA bulkhead was the only one in which fatigue had propagated; the fatigue area was about .03 inch long at each end of the overstress fracture.

Of the nine DC-10's with fractured flanges, only the accident aircraft had shims installed on the upper surface of the flange.

1.16.3 Stress Testing of the Pylon Aft Bulkhead

As a result of the discovery of the damaged upper flange on the accident aircraft, laboratory tests were conducted in an attempt to reproduce the 10-inch overload crack. The testing involved both static and dynamic loading with and without the .050-inch-thick shim installed on the flange. Static load tests conducted by American Airlines involved the use of a Tinnius Olsen universal test machine 2/ and a shimmed (.050-inch) spar web. The results showed that the flange cracked under a 6,400-lb load and when deflected .122 inch. The initial crack was 1.1 inch long. The crack progressed through the flange when the flange was deflected 0.2 inch after loading of 7,850 lbs; its length was 2.8 inches. Once the flange was penetrated, it required lighter loads to produce greater deflections. A crack 7.4 inches long was produced with a 5,175-lb load at a 0.6-inch deflection. At this point, the ends of the crack had disappeared under the spar web.

Additional static and dynamic load tests were conducted at McDonnell-Douglas. A test specimen, consisting of the aft bulkhead, connecting spar web, and a .050-inch shim installed between the bulkhead flange and the spar web, was used in one static test. A jackscrew was used to apply load to the specimen. Cracking began when the flange was deflected 0.1 inch with an applied load of about 6,900 lbs. At 11,500 lbs, the flange was deflected 0.2 inch, and the crack propagated to about 2 inches. Increasing the deflection to 0.6 inch lengthened the crack to 7.8 inches; however, the required load was only 8,600 lbs.

The evidence indicated that the maximum interference that would result from the insertion of the .050-inch-thick shim was .024 inch. The static tests conducted by American Airlines and McDonnell-Douglas showed that a crack would begin at a deflection of about 0.1 inch; thus, in the worst case, an additional deflection would be required to crack the flange.

2/ A machine used to apply precise and measurable amounts of stress to materials undergoing testing.

During dynamic testing, seven specimens were subjected to impact loads of varying energy levels and numbers of strikes. Specimens struck at high energy levels (6,000 inch-pounds) failed in unrelated modes. Specimens struck at low energy levels (1,500 to 2,500 inch-pounds) required seven to eight strikes to create an 8- to 10-inch-long crack. The total absorbed energy required to produce a 10-inch crack in an unshimmed specimen was 16,000 inch-pounds. The absorbed energy required to create a 10-inch crack in a shimmed specimen (.050-inch thick) was about 18,000 inch-pounds. In one test, a 10-inch crack was produced on an unshimmed specimen after two blows; the total absorbed energy was 5,200 inch-pounds.

In another test, conducted by American Airlines, an aft bulkhead, in which a 6-inch crack had been produced in the flange by forcing a simulated wing clevis vertically down on pylon web bolts, was subsequently subjected to a thrust load. With a thrust load of 11,625 pounds, the 6-inch crack extended to 10 inches, at which point the thrust load was relieved.

The major elements of the pylon structure were also examined to determine primary and fail-safe 3/ load paths. Normally the vertical and side forces, as well as torque or rolling moments in the plane of the bulkhead, are transmitted from the pylon structure through the spherical joints in the forward and aft bulkheads and into the wing structure. All of the thrust load from the pylon is intended to be transmitted through the thrust link.

The capability of the forward and aft bulkheads to serve as alternate fail-safe load paths in the event of a thrust link failure was assessed during the postaccident investigations. Therefore, in addition to the tests of the upper flange, a full-scale wing pylon test was conducted to evaluate load distributions and flexibility of the pylon-mounted bulkheads both with and without a thrust link installed.

The design gap between the forward and aft faces of the aft spherical bearing and the respective faces of the clevis is .080 inch. With this clearance, minimal thrust loads of about 600 lbs are experienced at the aft bulkhead. However, during the postaccident investigation, this gap was measured throughout the DC-10 fleet, and the smallest gap found was .047 inch. With that size gap and the engine at maximum thrust, thrust loads of about 6,650 lbs are experienced at the aft bulkhead; this load is still within the bulkhead's strength capability.

The failed thrust link tests showed that the thrust load was distributed between the aft and forward bulkheads—75 percent of the load (30,000 lbs) at the aft bulkhead and 25 percent at the forward bulkhead. The imposition of 75 percent of the thrust load on the aft bulkhead will shorten its service life. According to the evidence for the worst case, which is a DC-10-40 with the largest available engines, the estimated life of the aft bulkhead would be greater than 3,000 flight-

3/ Fail-safe means that in the event of a failure of a major element, the loads carried by that element are redistributed to another load path which can accommodate the load.

hours. The bulkhead of the DC-10 series 10, 30, and 40 aircraft are essentially identical. Further analysis based on the DC-10-10 thrust showed that an undamaged aft bulkhead would support the entire thrust load.

During the postaccident investigation, McDonnell-Douglas conducted flight tests to measure the wing pylon's relative deflection at the aft pylon mount and the stress created at selected nearby structural members throughout the normal flight regime. The flight regime investigated included, in part, taxi; takeoff including normal and rapid rotations; 2-G turns; moderate turbulence encounters; 2-G pullups, 0.2-G pushovers, landings, and rollouts; and the effects of maximum reverse thrust. The highest stresses measured on the aft bulkhead were less than 10 percent of the static strength of the material in the bulkhead.

Other tests were conducted at McDonnell-Douglas to determine the stress distribution and residual strength of the aft bulkhead under various load conditions with cracks in the forward flange. The aft bulkhead was mounted in a cantilevered structure that simulated the aft 3 ft of the pylon. Loads were applied to the bulkhead through the lug of the aft pylon at the wing attachment joint. The damage to the bulkhead was imposed by saw cuts, the ends of which were further cracked by the application of cyclic loads. Photo-stress and strain gage data were taken with the flange cracked 6 inches, 10.5 inches, and 13 inches; the latter condition was intended to replicate the crack and fatigue damage evident on the accident aircraft. It was determined that even a 6-inch crack would extend by fatigue progression with the application of cyclic loads representative of those encountered in service. The vertical and side loads representing those for a takeoff rotation with gusty crosswinds were applied to the bulkhead with the 13-inch crack without producing failure. A thrust component load was then added and increased to 9,000 lbs. at which time the bulkhead failed. The ends of the 13-inch crack, however, progressed to fastener holes, whereas the crack in the accident bulkhead did not. A theoretical analysis by a McDonnell-Douglas stress engineer showed that vertical and side loads alone could fail the bulkhead completely with a 13-inch preexistent crack in the forward flange.

During the reassessment of the fail-safe analysis of the aft bulkhead, the effect of a 6-inch fracture on the bulkhead's forward upper flange was further analyzed and tested. The crack location was similar to the locations of those found during postaccident inspections. The analysis and tests showed that the damaged structure could carry the fail-safe design loads for the worst case—the aircraft with the largest engine.

During ground operation of the aircraft--taxiing, landing, and takeoff rolls--the aft bulkhead is subjected to compression loads and the aft end of the pylon is forced upward. During rotation, the loading changes and the aft bulkhead is subjected to tension-type loads. Those loads were found to be significantly lower than the fail-safe design loads.

1.16.4 *Wind Tunnel and Simulator Tests*

The wind tunnel at the National Aeronautics and Space Administration's Langley Research Center was used to determine the aerodynamic characteristics

of a DC-10 wing with the left engine and pylon missing, left wing leading edge damaged, and the left wing's outboard leading edge slats retracted. In this configuration, the aircraft's stall speed, minimum control speeds with the critical engine inoperative (V_{MC}), and controllability were calculated. The effects that the loss of the No. 1 hydraulic system and the possible loss of the No. 3 hydraulic system would have on the aircraft's control authority were also investigated and calculated.

The DFDR data, aerodynamic data derived from wind tunnel tests, and the atmospheric conditions on the day of the accident were integrated into the Douglas Motion Base Simulator. The following conditions were simulated: (1) The separation of the No. 1 engine and pylon and the aerodynamic effects of the separation and resultant damage, such as changes in the aircraft's gross weight and lateral and longitudinal c.g.; (2) the uncommanded retraction of the left wing's outboard leading edge slats; (3) the loss of the No. 1 and No. 3 hydraulic systems; (4) the loss of power from the No. 1 a.c. electrical bus and resultant loss of the captain's flight instruments; and (4) both the loss and retention of the stall warning system and its stickshaker function.

The wind tunnel data for the damaged aircraft were correlated with the DFDR data so that the simulator data reflected those derived from Flight 191's DFDR. With the slats extended, the all-engine-operating stall speed was 124 KIAS; the asymmetric slat-retracted stall speed for the left wing was 159 KIAS; and the estimated wings-level V_{MC} for the damaged aircraft was 128 KIAS. With a 4° left bank-- a bank into the missing engine — 159 KIAS was the minimum speed at which directional control could be maintained with the engines operating at takeoff thrust.

Each of the thirteen pilots who participated in the simulation was thoroughly briefed on the flight profile of Flight 191. In the simulator the No. 1 engine and pylon assembly was programmed to separate at 10° of rotation on all takeoffs with simultaneous loss of the No. 1 hydraulic system. On some test runs the No. 3 hydraulic system was also programmed to fail. Generally, slats began to retract about 1 sec after the engine and pylon separated and were fully closed in about 2 sec. Some test runs were conducted with the slat retraction beginning 10 to 20 sec after the engine and pylon separated. Speed control guidance from the flight director was available for all runs, and the stickshaker, programmed for the slat-retracted-airspeed schedule, was operational on some runs.

During the tests, about 70 takeoffs and 2 simulated landings were conducted. In all cases where the pilots duplicated the control inputs and pitch attitudes shown on the Flight 191's DFDR, control of the aircraft was lost and Flight 191's flight profile was duplicated. Those pilots who attempted to track the flight director's pitch command bars also duplicated Flight 191's DFDR profile.

According to American Airline's procedures, the standard rate of rotation is between 3° to 4° per second, whereas Flight 191 rotated at only about 1.5° per second. In those simulations in which the standard rate was used, the aircraft lifted off at a lower airspeed, and the airspeed did not increase to the levels recorded by Flight 191's DFDR. The left roll began at 159 KIAS; however, because of the lesser amount of excess airspeed, the roll started below 100 ft a.g.l. In those cases where slat retraction was delayed, the left roll started at a higher altitude but its characteristics remained the same. In all cases, however, the roll began at 159 KIAS.

In many cases, the pilots, upon recognizing the start of the roll at a constant pitch attitude, lowered the nose, increased airspeed, recovered, and continued flight. The roll angles were less than 30°, and about 80 percent right rudder and 70 percent right-wing-down aileron were required for recovery. In those cases where the pilot attempted to regain the 14° pitch attitude commanded by the flight director command bars, the aircraft reentered the left roll.

On those test runs with an operative stickshaker programmed to begin at the slat-retracted-airspeed schedule, the stickshaker activated 7 sec after liftoff and the pilot flew the aircraft at the stickshaker boundary speed of 167 to 168 KIAS (V_2 + 15). Also, when V_2 + 10 was obtained and the pilot disregarded the pitch command bars, a stable climb was readily achieved. Attempts to duplicate the 1-sec interval of zero rudder displacement did not have any noticeable effect on the flight profile.

Based on the probable electrical configuration existing after the takeoff of Flight 191, pilots and test pilots who testified at the Safety Board's public hearing believed that the stall warning system and the slat disagreement warning light were inoperative. They stated that the flightcrew cannot see the No. 1 engine and left wing from the cockpit and, therefore, the first warning the flightcrew would have received of the stall was the beginning of the roll. Under these circumstances, none of these pilots believed that it was reasonable to expect the flightcrew of Flight 191 to react in the same manner as did the simulator pilots who were aware of Flight 191's profile and were able to recover from the stall.

The FAA conducted a second series of tests to determine the takeoff and landing characteristics of the DC-10 with an asymmetrical leading edge slat configuration. The slat configuration which existed on Flight 191 before impact was duplicated during about 84 simulated takeoffs and 28 simulated landings. Takeoffs were performed at both normal and slow rotation rates, at normal V_R speeds, at V_R -5 kn, and with thrust reduced to simulate a limiting weight condition during a second-segment climb.

The "slat disagree" light, takeoff warning system, and stall warning system were programmed to operate properly for both the normal and asymmetric outboard slat configuration.

Landings were performed at the maximum landing weight, 50° of flap, and a normal approach speed. The simulator was programmed so that a left outboard slat failure would cause the slat to fully retract at altitudes as low as 30 ft a.g.l. The FAA concluded that "The speed margins during the final portion of the landing approach are also very small; however, the landing situation is considered less critical since powered slat retraction from the landing configuration requires 18 seconds and an additional thrust is readily available to adjust the flight path." 4/

4/ Report to the Administrator on the Investigation of the Compliance of the DC-10 Series Aircraft with Type Certification Requirements under Asymmetric Slat Condition, July 9, 1979.

During these tests, none of the pilots experienced problems with aircraft controllability. In many of the test runs, the stickshaker activated at or just after liftoff, and the pilots altered the aircraft's attitude and airspeed in response to the warning. A loss of thrust from an engine during the takeoff roll was not simulated during any of the tests. Based on a study performed by the J. H. Wiggins Company 5/, the best estimates of the probabilities of an uncommanded slat retraction during takeoff ranged from one chance in one hundred million (1×10^{-8}) to two chances in a billion (2×10^{-9}) per flight.

1.17 Other Information

1.17.1 Air Carrier Maintenance Procedures

On May 31, 1975, and February 1, 1978, the McDonnell-Douglas issued DC-10 Service Bulletins 54-48 and 54-59, respectively. Both bulletins were issued to correct service-related unsatisfactory conditions. Service Bulletin 54-59 called for the replacement of the pylon forward bulkhead's upper and lower spherical bearings and contained procedures for accomplishing the maintenance. Compliance was recommended at the "operator's convenience."

Service Bulletin 54-48 called for the replacement of the pylon aft bulkhead's spherical bearing, and compliance with the modification was "optional, based on operator's experience." The procedures for accomplishing the modification contained the following note: "It is recommended that this procedure be accomplished during engine removal." The Service Bulletin later reiterated the recommendation and then stated, "The following instructions assume that engines 1 and 3 are removed." However, the vice president for maintenance and engineering, American Airlines, testified at the Safety Board's public hearing that the manufacturer's consideration for maintenance timing is not necessarily consistent with air carrier operations. For example, American Airlines' maintenance cannot forecast "with any great accuracy" when or where an engine would have to be changed. Since "it has to be scheduled," it would have been impractical to try to carry out the procedures of Service Bulletin 54-48 in that manner, and the aircraft would have to be scheduled to undergo the modification.

Service Bulletin 54-48 directed that the pylons were to be removed in accordance with the procedures contained in Chapter 54-00-00 of the DC-10 Maintenance Manual. Chapter 54-00-00 called for, first, removal of the engine and then removal of the pylon. The pylon alone weighs about 1,865 lbs. and its c.g. is located about 3 ft forward of the forward attachment points whereas the pylon and engine together weigh about 13,477 lbs. and the c.g. of the assembly is located about 9 ft forward of the forward attachment points. According to the manual, the sequence shown for the removal of the attach fittings was: The forward upper attach assembly, the forward lower attach assembly, the thrust link, and the aft bolt and bushing.

5/ Technical Report No. 79-1365, Estimating the Probability of Asymmetric Deployment of the Leading Edge Slat System of the DC-10 Aircraft, J. H. Wiggins Company.

American Airlines decided to comply with Service Bulletins 54-48 and 54-59 and to perform the work during a maintenance "C" check at its Tulsa maintenance facility. (See appendix C.) On July 28, 1978, American Airlines issued Engineering Change Order (ECO) R-2693 establishing the maintenance procedures for accomplishing the modifications contained in the service bulletins.

The ECO was developed from the company's experiences during modifications on four DC-10-30's during the spring and fall of 1977, at Los Angeles, California. American Airlines, in accordance with a contract with a foreign carrier, modified four of the foreign carrier's DC-10-30's. The carrier also requested that American Airlines perform the spherical bearing replacement program contained in Service Bulletins 54-48 and 54-59. While establishing the maintenance procedures for the four DC-10-30's, American's maintenance and engineering personnel evaluated the feasibility of raising and lowering the engine and pylon assembly as a single unit using a forklift-type supporting device. This technique would save about 200 man-hours per aircraft, but more importantly from a safety standpoint, it would reduce the number of disconnects (i.e., hydraulic and fuel lines, electrical cables, and wiring) from 79 to 27. American personnel knew that United Airlines was using an overhead hoist to lower and raise the engine and pylon assembly as a single unit.

American Airlines personnel contacted McDonnell-Douglas personnel about this procedure. According to the American Airlines' manager of production for the Boeing 747 and DC-10 in Tulsa, Oklahoma, who participated in the development of the maintenance procedures, a McDonnell-Douglas field service representative stated that McDonnell-Douglas did not know of any carrier that was removing the engine and pylon as single unit. He said that the field service representative conveyed concern "in reference to clearances to me." However, he assumed that these clearances involved those between the clevis and the fore and aft faces of the aft pylon bulkhead's spherical bearing.

The McDonnell-Douglas field service representative who was contacted by American's personnel stated that he conveyed American's intentions to his superiors. According to him, "Douglas would not encourage this procedure due to the element of risk involved in the remating of the combined engine and pylon assembly to the wing attach points" and that American Airlines' personnel were so advised.

McDonnell-Douglas does not have the authority to either approve or disapprove the maintenance procedures of its customers. American Airlines decided to lower the engine pylon assembly as a single unit and requested that McDonnell-Douglas provide it information concerning the c.g. of the engine and pylon, including the nose cowl and both fan cowl and core cowl thrust reversers, as a single unit. The single unit was to be lowered by a forklift. On March 31, 1977, the McDonnell-Douglas field service representative informed his company that American Airlines "proposes to drop the wing engines, pylon . . . as a single unit package directly on to an engine stand by means of a (forklift)" and then asked for the "C.G. of the pylon in the above described condition." On April 8, 1977, McDonnell-Douglas furnished the data to American.

The evidence showed that, during the time the procedure was in use, several McDonnell-Douglas employees saw the engine and pylon assembly after it was lowered from the wing; however, none of them observed either the actual mating, separating, raising, or lowering of the unit. Those who stated that they had seen the unit resting on the floor of a hangar also stated that they attached no significance to what they saw.

American Airlines used the newly developed removal method to modify the four foreign DC-10-30's. While working on the first aircraft, the maintenance personnel had difficulty removing the forward bulkhead's attach assemblies before removing the aft bearing bolt and bushing. They reversed the procedure and found that removing the aft bolt and bushing first expedited the removal of the forward attach assemblies and the thrust link. The reversed procedure was followed on the remaining three aircraft, and the modification program was completed. The fleet inspection conducted after the accident did not disclose any damage to the upper flanges on these four aircraft. However, the DC-10-30's aft bulkhead design affords more clearance between the bottom of the clevis and the upper spar web fasteners than the DC-10-10's design.

When the decision was made to modify American Airline's DC-10-10 fleet, the procedures used during the DC-10-30 modification program were adopted and incorporated in ECO R-2693. A Hyster forklift, Model 460B, American Airlines No. 3145, was used to raise and lower the engine and pylon assembly. The forklift has a design load capacity of 42,500 lb. In addition to vertical movement, the lifting forks can be moved in several directions: They can be yawed from left to right, tilted from left to right, tilted forward and aft, or moved laterally by moving the mast in the desired direction. A new c.g. for the DC-10-10's engine and pylon assembly was computed by American Airlines and instructions for centering the forklift at the c.g. were incorporated in the ECO. The operator was directed to insert the forks into an engine shipping stand and attach the supported stand to the engine. The ECO stated, "Adjust the engine support adapter aft so that the centerline of the lifting forks are centered with the center hinge on the (engine's) thrust reverser." The lifting forks are 5 ft apart. There was no mark on the forklift denoting the midpoint between the forks; therefore, alignment was a visual estimate. (See figure 13.)

The engine shipping stand, which can be used to support either a JT9D or CF6-6D engine, was used to support the engine and pylon assembly on the lifting forks. The stand can be adjusted for the different c.g.'s of the two engines, which are denoted by an arrow. The stand has a movable top cradle to which the engine is affixed; the cradle can be moved about 12 inches horizontally. There is also an arrow on the cradle's frame. The arrow on the frame of the cradle must be aligned with the arrow denoting the type engine to be loaded before the engine is placed on the cradle. Eight clamps secure the cradle in position on the stand. However, the cradle can be moved on the stand after the engine has been affixed to it.

American Airlines' maintenance personnel testified about their experiences with the forklift while handling the engine and pylon assembly. Directions were transmitted to the lift operator either by voice, hand signals, or both. The testimony varied regarding the capability to raise or lower the lifting forks a finite distance. One mechanic said it could be limited to .001 inch; the estimates of others ranged from .25 to .06 inch.

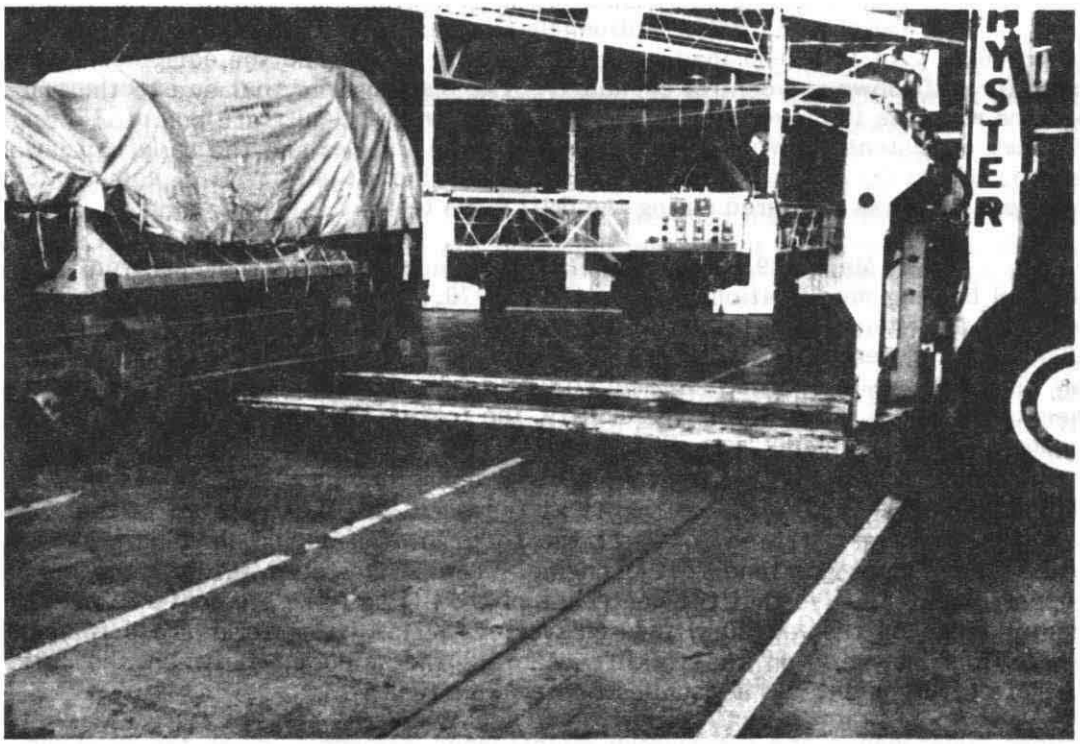


Figure 13. View of forklift and engine stand.

When the full weight of the engine and pylon assembly was on the lifting forks, the pressure gauge reading was 18,000 lbs. Maintenance personnel stated that a 2,000-lb to 3,000-lb pressure bleedoff on the pressure gauge was common; however, they all stated that the lifting forks did not move. Supervisory personnel stated that it was normal for the gauge reading to bleed off 2,000 to 3,000 lbs during a 15-min period without any perceptible load movement. The load remains fixed because of the frictional load on the mast and rollers. Although mechanics testified that the load did not move, they also said that they would manipulate the controls to restore the original reading on the pressure gauge. One mechanic stated that the pylon and engine assembly would "jump" as lowering began. He said the "jerking" motion moved the forks about 1 or 2 inches.

On October 5, 1979, McDonnell-Douglas tested the capability of its Hyster 460B. An 18,000-lb load was placed on the forks, and the equipment was tested for drift down and control capability. The tests showed that an experienced operator was able to move the load in both directions vertically in steps of .187 to .250 inch consistently. When the load was stopped the peak dynamic deflections were (+) .03 to (+) .06 inch about the final rest value. A sink rate of about 1.25 inches per hour was measured during the drift down test.

From March 29 through 31, 1979, the accident aircraft underwent the spherical bearing modification. On April 19, 1979, the forklift's maintenance log contained a writeup which noted, in part, "trouble shooting, forks creeping down under load." There was no record that any corrective action was taken. On May 17, 1979, the log showed "Inspect lift cylinder--per Engineering." There was no record of any findings. On June 20, 1979, the forklift was tested for drift down. An engine-pylon assembly was placed on the lifting forks, and the forks drifted down 1 inch in 30 min. A lift cylinder check valve was found to be defective and was replaced.

The ECO's procedures for detaching the pylon from the wing were as follows: Item F of the ECO called for the removal of the lower attach plug and attaching parts; item G called for removal of the upper attach plug and attaching parts; item H called for removal of the thrust link; and item I--called for removal of the aft bolt and bushing. The ECO did not caution or advise that items F through I must be performed in the sequence listed. According to American Airlines maintenance and supervisory maintenance personnel, since the ECO did not contain such advice, it did not require that items F through I be performed in sequential order. Rather, it merely provided a checklist and signoff sheet to insure that all the steps were performed. Consequently, maintenance personnel saw no harm in performing the modification by first removing the aft spherical bearing's bolt and bushing. Engineering personnel who drafted the ECO were not informed formally of the difficulties experienced in removing the fittings as prescribed.

Mechanics and the inspector who performed the spherical bearing modification on the accident aircraft recounted the operation for the Safety Board. The midnight shift started the modification and removed the aft spherical bearing's bolt and bushing before going off duty on March 30. When the day shift reported for duty, two of the mechanics saw the upper lug of the aft bulkhead come in contact with the bolts attaching the clevis to the wing. These bolts are located at the top of the clevis. The forklift's engine was running at the time, and the

pressure gauge reading was 18,000 lbs. When the crew could not remove the forward attach assemblies, they discovered that the engine stand was misaligned. The clamps holding the cradle to the stand were loosened, and the lifting forks and engine stand were shifted to the left--forward on the engine--until the cradle was properly aligned on the stand, The clamps were then affixed. According to one mechanic, the stand was moved forward about 12 inches. After the stand was realigned, the forward upper and lower attach assemblies were removed, and the engine and pylon assembly was lowered to the hangar floor.

The testimony of the mechanics disclosed that the mechanics' training for this modification was limited to on-the-job training. The inspector had not received any training with regard to this particular modification.

The work cards used to accomplish the modification on the accident aircraft were examined. The inspector's signoff blocks on the ECO's work cards did not contain any requirement for the inspector to inspect the forward or aft attach assemblies after the pylon and engine had been reinstalled on the wing. The work cards included in the ECO showed that after the inspector cleared the pylon for installation, his only inspection requirements were to inspect the connections for integrity and to check for fuel and hydraulic leaks. The work cards also disclosed that there was a nick on the top surface of the pylon aft bulkhead's attach lug, and some of the mechanics recalled seeing the nick.

The inspector stated that chronic problems in the maintenance procedures should be reported on a significant item form. This form is then channeled through maintenance supervision to engineering for action. He said that he thought that the out-of-sequence performance of the tasks in an ECO should be reported to those who formulate the ECO's.

In summary, an overall assessment of the manner in which American Airlines' Engineering developed and then monitored the two ECO's used to replace the pylon's spherical bearings showed they had evaluated the capabilities of the forklift before the decision was made to use the equipment. The engineer who wrote the procedures knew that the forklift was capable of applying high forces. He believed that the movement of the lifting forks could be controlled within "very small fractions" of an inch, but he did not know the resultant rate of movement of these forks in response to a control input. However, since the maintenance personnel were familiar with the forklift, he believed that its use would be more suitable for "our operation."

According to the engineer the procedures of the ECO's and the capabilities of the forklift were analyzed for safety of operation and personnel informally. However, they did not use or perform a formal fault analysis to evaluate the effect on the structure that might result from either personnel error or equipment malfunction. Procedures of this nature, according to the engineer, had never been used to evaluate ECO's. Members of the engineering department observed the prototype procedure on the first two DC-10-30's. However, they only observed the lowering and raising of the engine and pylon as a single unit. They did not witness the removal of the wing to pylon attach assemblies; consequently they were not aware of the difficulties that were encountered, and the subsequent departure from the sequence contained in the ECO.

The maintenance procedures used by Continental Airlines to accomplish SB 54-48 were similar to those of American Airlines. The same type forklift was used to raise and lower the pylon and engine assembly.

On December 19, 1978, the upper flange of the No. 1 pylon aft bulkhead on Continental Airlines DC-10-10, N68041, sustained a crack which penetrated the flange. The upper and lower forward attach assemblies had been removed, and the aft spherical bearing's bolt and bushing had been removed and a pin inserted in its place. When the pin was removed, the aft end of the pylon moved up slightly and a "loud pop described as a pistol shot" was heard. The fracture was discovered, the upper flange repaired, and the aircraft was returned to service.

On February 22, 1979, the upper flange of the No. 3 pylon aft bulkhead in Continental Airlines DC-10-10, N68049, sustained a crack which penetrated the flange. In this case the pylon had been disconnected and the lead mechanic was attempting to clear the aft bulkhead lug from the clevis. He instructed the forklift operator to raise the nose of the engine in order to lower the aft end of the pylon. The forklift operator either misunderstood or inadvertently moved the wrong lever and lowered the nose. The aft end of the pylon was raised with the same results and noise effects described above. Continental Airlines' investigation concluded that both mishaps were maintenance errors and neither was reported to the FAA.

The forklift was checked for drift down after the December mishap and "nothing was found." Several months later the unit was rechecked. Downward drift was found, the malfunction was corrected, and no further difficulties were encountered.

During the postaccident investigation, the maintenance procedures of all United States carriers operating DC-10 series aircrafts were inspected. The evidence disclosed that United States carriers had removed and reinstalled 175 pylon and engine assemblies. Eighty-eight of these operations involved the lowering and raising of the pylon and engine as a single unit. Of these 88, 12 were lowered and raised with an overhead crane. The remaining 76 were lowered and raised with a forklift. The nine situations wherein impact damage was sustained and cracks found involved the use of the forklift.

1.17.2 Federal Aviation Administration Reporting and Surveillance Procedures

Air carrier reporting requirements are established in 14 CFR Part 121, and are basically contained in two regulations. 14 CFR 121.703 establishes the mechanical reliability report (MRR) system. The regulation requires a certificate holder to report "the occurrence or detection of each failure, malfunction, or defect concerning . . . ." The regulation contains 16 paragraphs setting forth the conditions that must be reported. 14 CFR 121.703(14) requires the carrier to report "Aircraft structure requiring major repair" and paragraph (15) requires the carrier to report "cracks, permanent deformation, or corrosion of aircraft structures, if more than the maximum acceptable to the manufacturer or the FAA." According to the FAA, the MRR system is, for the most part, limited to service-related problems and to failures and malfunctions which have occurred after the aircraft's engines are started with the intent for flight and while they are running. In response to a question as to whether paragraph (15) would apply to the

December 1978, and February 1979, upper flange cracks at Continental Airlines, a FAA air carrier maintenance specialist stated that historically and traditionally the MRR procedures have always dealt with service-related problems. He said that under the MRR concept "we would not consider it because it was not a service related problem."

14 CFR 121.707, Alteration and Repair Reports reads as follows:

(a) Each certificate holder shall, promptly upon its completion, prepare a report of each major alteration or major repair of an airframe, aircraft engine, propeller, or appliance of an aircraft operated by it.

(b) The certificate holder shall submit a copy of each report of a major alteration to, and shall keep a copy of each report of a major repair available for inspection by, the representative of the Administrator assigned to it.

The authority for an air carrier to perform maintenance is derived from several sources. Pursuant to the provisions of 14 CFR 21, Subpart M, an air carrier may be certified by the FAA as a Designated Alteration Station (DAS), as were American and Continental Airlines. In accordance with this certification, either carrier could issue supplemental type certificates and perform its own alterations without prior FAA approval; however, the required reports must be submitted to the FAA.

14 CFR 121.379 also contains authorization for a Part 121 certificate holder to perform maintenance and alterations. This section reads, in part, as follows:

"(a) A certificate holder may perform, or it may make arrangements with other persons to perform maintenance, preventive maintenance, and alterations as provided in its continuous airworthiness maintenance program and its maintenance manual. . . .

"(b) A certificate holder may approve any aircraft, airframe, aircraft engine, propeller, or appliance for return to service after maintenance, preventive maintenance, or alternations that are performed under paragraph(a) of this section. However, in the case of a major repair or alteration, the work must have been done in accordance with technical data approved by the Administrator."

The investigation showed that there were large differences in the interpretation of what constituted a major alteration or repair despite the guidelines contained in the Federal regulations. 14 CFR 1.1 defines a major repair and alteration as follows:

"A major alteration means an alteration not listed in the aircraft, aircraft engine, or propeller specifications--(1) That might appreciably affect weight, balance, structural strength, performance, powerplant operation, flight characteristics or other qualities affecting airworthiness; or (2) That is not done according to accepted practices or cannot be done by elementary operations.

> "'Major Repair' means a repair: (1) That, if improperly done might appreciably affect weight, balance, structural strength, performance, powerplant operation, flight characteristics, or other qualities affecting airworthiness; or (2) That is not done according to accepted practices or cannot be done by elementary operation."

The FAA air carrier maintenance specialist stated that the classification of major alteration or repair related to the requirement that either or both be accomplished in accordance with approved data. It is a method of protecting the type certificate design and of assuring that the repair or alteration does not change or modify a design feature.

Continental Airlines' principal maintenance inspector stated that there are no "clear cut rules" for interpreting the regulation. "It has been argumentative for 30 years that I know (sic) it." Although it was his opinion that the major part of the bulkhead was a structurally significant item, he did not consider the upper flange part of the bulkhead.

The FAA team investigating maintenance and airworthiness procedures after the accident found that FAA regulations and guidance did not adequately define what constitutes a major repair. The team found that the repairs made to five pylons, including the two upper flanges at Continental Airlines, constituted major repairs since critical structure was involved. Therefore, the team concluded these repairs should have been submitted to the FAA for approval. 6/

The FAA principal maintenance inspectors are responsible for the surveillance of the maintenance activities and procedures of those air carriers assigned to their office. The principal inspector for Continental Airlines was not aware of the cracks sustained in the upper flanges of the two Continental aircraft during the modification procedure, nor did he know when the carrier began the modifications contained in Service Bulletins 54-48 and 54-59.

The principal maintenance inspector at American Airlines' Tulsa Maintenance Base was also the chief of the Tulsa Air Carrier District Office, and had served 7 years as chief. However, he had been principal inspector for American Airlines since January 15, 1979. The principal maintenance inspector did not know that American Airlines was removing the pylon and engine assembly as a single unit until May 30, 1979. In accordance with a request from his office sometime before May 23, 1977, American Airlines had been requested to revise its ECO distribution to the Tulsa Air Carrier District Office to "include cover sheets only, without the detailed technical data." Thus, the FAA received only the cover sheet of ECO R-2693. The material containing the maintenance procedures was retained by the carrier, and the Tulsa Air Carrier District Office did not conduct any checks on the pylon maintenance.

6/ Report to the Administrator of the Federal Aviation Administration in the Matter of Maintenance and Airworthiness Procedures Concerning DC-10 Aircraft Operated By American Airlines, Continental Airlines, and six other U.S. Air Carriers.

The cover sheet of ECO R-2693 classified the repairs as minor. The principal inspector said that the cover sheet also contained the FAA-approved Service Bulletins 54-48 and 54-59. Therefore, he had no reason to either doubt the classification or the carrier's capability to carry out the repair. In his opinion, there was no reason to expend manpower in surveillance of a minor repair.

Evidence developed during the investigation showed that FAA approval of a service bulletin indicates to the operator that the change in design included in the bulletin has been approved by the FAA, thereby relieving the operator of the necessity of obtaining his own design approval. However, the FAA approval does not apply to the maintenance procedures incorporated in the service bulletin.

1.17.3 DC-10 Certification

The DC-10's pylon structure, flight controls, hydraulic system, and electrical system were certificated in accordance with the applicable provisions of 14 CFR Part 25 effective February 1, 1965, as amended, and Special Condition No. 25-18-WE-7, January 7, 1970, as amended. (See appendix E.)

Special Condition No. 25-18-WE-7, Docket No. 10058, was issued pursuant to 14 CFR 21.16 because the airworthiness regulations of Part 25 did not contain adequate or appropriate safety standards for the aircraft because of a novel or unusual design feature. In the case of the DC-10, this feature was the fully powered flight control system.

The function of assessing compliance with certain aspects of the type certification was delegated to FAA Designated Engineering Representatives who were employed by McDonnell-Douglas. Such representatives are designated by the FAA to represent the Administrator pursuant to Section 314 of the Federal Aviation Act of 1958 and 14 CFR 183.29. According to FAA and McDonnell-Douglas witnesses, the workload involved in the certification process far exceeds the FAA's manpower resources.

The chief of the FAA's Western Region Aircraft Engineering Division stated that during the type certification process the review of the basic data and the most critical tests are reserved to the FAA itself. The fault analysis data are reviewed and approved by FAA engineering personnel. He also said that little delegation is done in the flight test area. The chief of the FAA's Western Region Flight Test Branch stated that the DC-10's type certification required 500 hrs of flight testing, and 90 percent of that time was flown by FAA test pilots.

The principle underlying the regulations concerning the certification the aircraft's systems was redundancy. This principle contemplates that, while each critical component of a system is required to perform functions within the design envelope of the aircraft, its failure will nevertheless be assumed. Accordingly, appropriate analyses and tests are required to insure that sufficient redundancy exists so that after a single failure of any component or element its functions will be distributed to other components capable of assuming them safely.

The criteria for the certification of the aircraft's pylon and its components were contained in 14 CFR 25.571, "Fatigue Evaluation of Flight Structure". (See appendix E.) This regulation required the manufacturer to show, by analysis, tests, or both, that those parts of the structure whose failure could result in catastrophic failure of the aircraft would be able to withstand the repeated loads of variable magnitude expected in flight, that catastrophic failure or excessive structural deformation that could adversely affect the flight characteristics of the aircraft are not probable after fatigue failure or obvious failure of a single principal structural element, and that after this type of failure of a single principal structural element, the remaining structure must be able to provide an alternate load path. The regulation only required that fatigue damage be evaluated. The chief of the FAA's Western Region Aircraft Engineering Division testified that under normal loading there was "extremely low stress" on the upper flange and "the possibility of fatigue was believed to be extremely low, low enough that you would not consider fatigue failure.

Because all flight controls were hydraulically actuated and the basic regulations did not cover this configuration, Special Condition No. 25-18-WE-7 was formulated. However, the trailing edge flap and leading edge slat systems were certified under the basic regulations.

The leading edge slat system was certified in accordance with 14 CFR 25.671--general control system requirements, 14 CFR 25.675--control system stops, 14 CFR 25.685--detailed design requirements for flight control systems, and 14 CFR 25.689--cable system design. The chief program engineer at McDonnell-Douglas said that the flap control requirements of 14 CFR 25.701(a) were also applied to the slats. Paragraph (a) states:

> "The motion on the flaps on opposite sides of the plane of symmetry must be synchronized unless the aircraft has safe characteristics with the flaps retracted on one side and extended on the other."

Since the left and right inboard slats are controlled by a single valve and actuated by a common drum and the left and right outboard slats receive their command from mechanically linked control valves which are "slaved" to the inboard slats by the followup cable, the synchronization requirement was satisfied. However, since the cable drum actuating mechanisms of the left and right outboard slats were independent of each other, the possibility existed that one outboard slat might fail to respond to a commanded movement. Therefore, the safe flight characteristics of the aircraft with asymmetrical outboard slats were demonstrated by test flight. These flight characteristics were investigated within an airspeed range bounded by the limiting airspeed for the takeoff slat positions --260 kns--and the stall warning speed; the flight test did not investigate these characteristics under takeoff conditions. In addition, a slat disagree warning light system was installed which, when illuminated, indicated that the slat handle and slat position disagree, or the slats are in transit, or the slats have been extended automatically.

The program engineer stated that the commanded slat position is held by trapped fluid in the actuating cylinder, and that no consideration was given to an alternate locking mechanism. The slats' hydraulic lines and followup cables

were routed as close as possible to primary structure for protection; however, routing them behind the wing's front spar was not considered because of interference with other systems.

The branch chief of the Reliability and Safety Engineering Organization of the Douglas Aircraft Company described the failure mode and effects analysis (FMEA) and fault analysis. The witness indicated that the FMEA was a basic working document in which rational failure modes were postulated and analyzed; vendors and subcontractors were requested to perform similar analyses on equipment they supplied to McDonnell-Douglas. Previous design and service experience was incorporated in the initial DC-10-10's FMEA's and analyses were modified as the design progressed. The FMEA's were synthesized to make fault analyses, which were system-oriented summary documents submitted to the FAA to satisfy 14 CFR 25.1309. The FAA could have requested and could have reviewed the FMEA's.

The basic regulations under which the slats were certified did not require accountability for multiple failures. The slat fault analysis submitted to the FAA listed 11 faults or failures, all of which were correctable by the flightcrew. However, one multiple failure--erroneous motion transmitted to the right-hand outboard slats and an engine failure on the appropriate side--was considered by McDonnell-Douglas in its FMEA. The FMEA noted that the "failure increases the amount of yaw but would be critical only under the most adverse flight or takeoff conditions. The probability of both failures occurring is less than 1×10^{-10}." The evidence indicated that this FMEA was not given to the FAA formally but was available for review.

Special Condition No. 25-18-WE-7 requires the applicant to show that the aircraft is capable of continued flight and landing after "any combination of failures not shown to be extremely improbable." According to FAA witnesses, the definition for extremely improbable that they have been using and have been accepting for a number of years is one chance in a billion, or 1×10^{-9}.

The regulation, 14 CFR 25.207, requires that "Stall warning with sufficient margin to prevent inadvertent stalling with the flaps and landing gear in any normal position must be clear and distinctive to the pilot in straight and turning flight." The warning can be furnished through the inherent aerodynamic qualities of the aircraft or by a mechanical or electronic device. A visual warning device is unacceptable. The warning must begin at a speed exceeding the stall speed or the minimum speed demonstrated ". . . .by seven percent or at any lesser margin if the stall warning has enough clarity and duration, distinctiveness, or similar properties." The flight testing of the DC-10 disclosed that the inherent aerodynamic stall warning exceeded the required regulatory margin in all flap configurations until the landing flap configuration (50°) was reached. According to the chief of the FAA's Flight Test Branch, with 50° flaps the stall buffet still precedes stall onset, "but it occurs quite close, within just a few knots of the aerodynamic stall." Since the margin did not meet the regulatory criteria, a stall warning system was installed.

The initial DC-10 design incorporated the left (No. 1) and right (No. 2) autothrottle speed computers (AT/SC) as stall warning computers. The No. 1 and No. 2 AT/SC's were powered by the No. 1 and No. 3 a.c. buses, respectively. The

No. 1 AT/SC received inputs from the left inboard flap position transmitter, from a position sensor on the left outboard slat section, and the left angle-of-attack sensor. The No. 2 AT/SC received its inputs from counterpart sensors and components on the right side of the aircraft. The stickshaker motor was mounted on the captain's control column and was powered by the No. 1 d.c. bus. A stall signal from either computer would actuate the stickshaker motor. The design contained provisions for a second stickshaker motor to be mounted on the first officer's control column; however, the second stickshaker was a customer designated option. The accident aircraft's stall warning system did not incorporate the second stickshaker described above.

The December 1, 1978, revision of 14 CFR 25.571 retitled the regulation "Damage-Tolerance and Fatigue Evaluation of Structure." The fail-safe evaluation must now include damage modes due to fatigue, corrosion, and accidental damage. According to the manufacturer, the consideration for accidental damage was limited to damage which can be inflicted during routine maintenance and aircraft servicing.

The FAA's Aircraft Engineering Division chief also stated that while the recertification process disclosed a deficiency in design data on file with the FAA it did not disclose any deficiency in the pylon's design. In some cases, the manufacturer had the data on file. In one instance, the data concerning the alternate load paths for thrust loads following a thrust-link failure were questioned. The manufacturer's analysis assumed the loads would be carried by the forward bulkhead. The manufacturer also stated that the thrust loads could be carried out by the aft bulkhead. The FAA asked McDonnell-Douglas to substantiate this claim, and they did so successfully.

As a result of the postaccident simulator tests, an AD was issued which required, as a condition for reinstatement of the type certificate, that the aircraft be operated either with both AT/SC's installed and operating, or with a modified single AT/SC that would receive slat information from both sides of the aircraft. (See appendix F).

On July 30, 1979, a Notice of Proposed Rule Making (NPRM), docket No. 79WE-17AD, was issued. (See appendix F.) The NPRM contained an AD which will require that the stall warning system incorporate two AT/SC's and two stickshaker motors, and that the AT/SC's be modified to receive position information from both outboard wing leading edge slat groups.

1.17.4 DC-10 Maintenance and Inspection Programs

During the investigation, the development of the DC-10 maintenance program was studied to determine the methods used to establish the aircraft's maintenance program and the inspection requirements for the wing pylons. The program guidelines were embodied in the "Airline Manufacturer Maintenance Program Planning Document, MSG-2." The document was formulated by a working group composed of representatives of user air carriers, McDonnell-Douglas, and one or more FAA observers. The document was then submitted to the FAA Maintenance Review Board where FAA observers and engineers met to evaluate the proposals. The review board issued a report which prescribed the minimum maintenance program for DC-10 operators and required a review of the specific work programs of each operator by its FAA principal maintenance inspector to assure conformance with the program.

When an aircraft is delivered to an operator, the manufacturer must, by regulation, furnish the operator with a maintenance manual (14 CFR 25.1529). The manual must contain the essential information and procedures necessary to maintain the aircraft.

The maintenance programs for modern aircraft are comprised generally of three primary maintenance processes known throughout the industry as "hard-time," "condition-monitoring," and "on-condition." Hard-time is a preventive maintenance process which requires that an appliance or part be overhauled or replaced after a specific period of service. This process is generally applied to parts which are subject to predictable wear, such as engines or engine components. Condition-monitoring is a process which applies to components, the output of which can be monitored to detect degradation in performance indicating that the maintenance is required. When applying the condition-monitoring process, the potential effect of an unpredicted failure of the part is also considered.

The airworthiness of most of the structural elements of the aircraft is maintained by the on-condition maintenance process. This process requires that a part be periodically inspected against some physical standard to determine whether it can continue in service. Thus, the maintenance program established for the aircraft includes specified inspection requirements for each structural element. The inspection interval depends upon an analysis which considers the susceptibility of the part to fatigue damage, corrosion, and crack propagation. The degree of redundancy and the accessibility for inspection are also considered.

The on-condition process also incorporates the principle that similar parts behave in similar ways. Thus, if a part is analyzed to be relatively resistant to damage throughout the anticipated life span of the aircraft, an inspection of that part on every aircraft--a 100-percent inspection--may not be required; the part will be placed in a sampling inspection program and a statistically representative sample of the parts on the entire fleet of aircraft will be inspected. If a problem is detected during the sampling inspection program, the FAA's service difficulty reporting program incorporates the mechanism whereby revised inspection requirements can be evaluated and levied on the operators for application to the entire fleet of aircraft. The on-condition maintenance program, thus, is intended to be a conservative method to verify the design resistance to fatigue or corrosion damage during the aircraft's service life. However, the maintenance programs are not designed to detect damage resulting from improper manufacturing processes or maintenance.

During the investigation, the Safety Board examined closely the sampling inspection program for the wing pylon. The program, sampling base, and inspection frequency were based upon factors, including projected aircraft life as well as structurally significant items and their resistance to fatigue and corrosion.

The maintenance document (MSG-2) defined structurally significant items as "those local areas of primary structure which are judged by the manufacturer to be relatively the most important from a fatigue or corrosion vulnerability or from a failure defect standpoint," and it required that these items be classified as to relative importance. The classification and ratings of these items were based upon the fatigue, corrosion, and crack-propagation resistance properties of the structure. These properties were analyzed on the basis of fatigue testing, special tests for crack growth rates, and the company's previous experience with the aircraft structure.

The structurally significant items with a classification or rating number of 1 or 2 (indicative of a lower overall level of structural integrity) would probably be placed on a 100-percent inspection program. The 100-percent program would require that these items be inspected on every aircraft at an interval which is determined by testing and analysis. Structures classified as 3 or 4 would probably be sampled. The sampling program required the inspection of some structurally significant items on only a specified fraction of an operator's fleet and at a predetermined interval.

The initial DC-10 program required 100-percent inspection for some items and placed others on a fractional sampling program. The inspection frequency for some items on the 100-percent program was based on their classification. Structurally significant items (SSI), classified as a Class-1 SSI, were to be inspected on all aircraft every 4,000 hrs, Class-2 SSI's every 8,000 hrs, and a Class-5 SSI every 20,000 hrs. In the fractional sampling program, only a certain proportion of a carrier's aircraft was to be inspected to monitor the condition of a structurally significant item. Thus, only 1/5 of a carrier's aircraft population was to be inspected at a 20,000-hr interval to monitor a Class-1 SSI, whereas 1/12 of its aircraft population was to be inspected at a similar interval to monitor a Class-5 SSI. For example, under the sampling program, the upper attach lug of the pylon aft bulkhead was on a 100-percent inspection program, while the aft bulkhead's upper flange and other portions of the bulkhead were on a fractional sampling program. The upper attach lug is designed to fail in the event of a wheels-up landing and thus prevent fuel tank rupture; accordingly, the lug was not overdesigned and is subjected to significant stresses which places it in a class requiring 100-percent inspection every 4,000 hrs. In contrast, the rest of the bulkhead is subjected to relatively low stresses; therefore, it is considered to be less susceptible to difficulty in service and suitable for sample-type inspection.

1.17.5 Manufacturer's Service Bulletin and Customer Service Programs

The FAA's service difficulty reports and McDonnell-Douglas service bulletins were reviewed to determine if any chronic difficulties related to aft bulkhead cracking had existed before the accident. The service difficulty reports indicated that some problems existed with wing spherical bearing attach fittings. These problems were not anticipated during design and did not develop until the aircraft was placed into service. As a result, programs were launched to replace the old spherical bearings with stronger and more efficient bearings through Service Bulletin 54-48 and 54-59.

McDonnell-Douglas maintains a customer support program. Under this program, the company maintains field service representatives at the operators' maintenance facilities and receives reports from operators concerning service difficulties encountered by its aircraft.

During December 1978, when Continental Airlines cracked the forward flange of an aft bulkhead during its bearing modification program, McDonnell-Douglas provided the operator with a engineer product specialist to assist it in repairing the flange. The product engineer specialist testified that he was responsible for investigating, analyzing, and interpreting customer reports regarding unsatisfactory performance and service failure of the aircraft structure.

He was also responsible for supplying any necessary corrective procedures. At the Safety Board's public hearing the engineer specialist testified that he did not see the pylon and engine assemblies raised or lowered, that he assisted the carrier in making the required repairs, and that he was told that the carrier "cracked the part while lowering the pylon. And that was the extent of the discussion."

According to the engineer, about 1 week later he wrote a short paragraph describing the problem and its disposition for inclusion into a company Operational Occurrences Report. This was published on January 5, 1979, as part of Report No. 10-7901 and read as follows:

> "An operator has reported a case of damage to the wing pylon aft monoball (spherical) bearing support bulkhead, P/N AUB7002-1. This apparently occurred when the pylon shifted while it was being lowered. The aft end of the pylon rotated up, and the forward lug of the wing clevis fitting contacted the upper horizontal flange of the support bulkhead. The flange on the support bulkhead was sheared off for most of its length; necessitating removal of the support bulkhead from the pylon for repairs."

Operational Occurrence Reports are distributed to all DC-10 operators. American Airlines did not recall receiving this Operational Occurrence Report, but Continental Airlines found it in its service library after the accident. The Operational Occurrence Report contained reports concerning all types of mishaps, system malfunctions, and structural defects that the manufacturer believed would be of interest to his customers. The report which contained the description of the bulkhead damage also contained reports of an air conditioning pack malfunction, a lightning strike, collapse of a passenger loading stand, and a flight attendant injury suffered in the galley cart lift.

14 CFR 21.3 establishes the responsibility of the holder of a type certificate to report failures, malfunctions, or defects to the FAA. The regulation requires a certificate holder to report any defect in any product or part it manufactures and that it has determined resulted in any of the occurrences set forth in the regulation. The primary structural defects the certificate holder is required to report are limited to those caused by "any autogenous condition (fatigue, understrength, corrosion, etc.)." Further, 14 CFR 21.3 (d)(i) states that the reporting requirements do not apply to failures, malfunctions, or defects that the certificate holder "determines were caused by improper maintenance."

1.17.6 Manufacturer's Production Line Procedures

The production line procedures of the facilities producing the wing pylon assembly were investigated, including the installation of shims on the upper surface of the horizontal flange on the accident aircraft. According to the McDonnell-Douglas' Vice President for Quality Assurance, the .063-inch-thick shims installed on the upper shoulders of the upper flange were standard shims. He said that these shims can be installed any place they are needed to reduce a clearance. No approval is needed since the procedure is authorized by Douglas Process Standard 2.70.2 (DPS 2.70.2).

The 10-inch-long, .050-inch-thick shim installed on the accident aircraft was not a standard shim and, according to McDonnell-Douglas engineers who testified at the Safety Board's public hearing, written authorization was required to use it. Such an authorization is processed through the company's engineering liaison group and reviewed by stress liaison personnel of the structural analysis group. Rejection and Disposition Item AO81757 had been issued authorizing the insertion of the shim, and had been signed by an engineer in the liaison group. Although a McDonnell-Douglas engineer assumed that the proper stress analysis had been performed before the issuance of the Rejection and Disposition Item, there was no signature to indicate specifically that the analysis had been done, nor was space provided for such a signature.

The evidence disclosed that 23 pylons were placed into service with shims on the top of the upper flange. The clearance problem on the upper flange began with fuselage No. 15 and continued through fuselage No. 36 (the accident aircraft was fuselage No. 22). A McDonnell-Douglas investigation disclosed that the clearance problem was the result of a tooling malfunction, and it was resolved by repositioning locator pins on the tooling jigs.

In October 1974, the pylon production line was transferred from McDonnell-Douglas' Santa Monica, California, location to the Huntington Beach, California, facility. The transfer was made at fuselage No. 208. During an inspection conducted after the accident, 31 aircraft were found to have had wing pylons with loose, failed, or missing fasteners. Fifteen of these aircraft were between fuselage No. 170 and 208. Six of these 15 aircraft had more than 5 loose or missing fasteners. Of the other 16 aircraft, 1 had 7 and another had 5 loose or missing fasteners; the remaining 14 aircraft had less than 5 loose or missing fasteners. McDonnell-Douglas personnel believed that one of the causes of this production breakdown was the effect the impending transfer of the production line had upon worker experience, morale, and productivity.

The investigation of the upper spar web cracks and fasteners found on United Airlines DC-10, N1827U, fuselage No. 196, also showed that its problems probably were traceable to production line procedures at McDonnell-Douglas. The damage on the United Airlines DC-10 was limited to the cracking of the upper spar web and failure of 26 fasteners. There was no damage to the aft bulkhead flange. An examination of the aircraft's history showed that it had not been exposed to any hard landings; however, it had experienced an engine failure and had been subjected to vibrations resulting from the windmilling of an unbalanced engine during 1 hr 20 min of flight.

Engine vibration testing was conducted at the General Electric facility at Peebles, Ohio, to investigate the possibility that a significant imbalance accompanied with windmilling for 80 min was a possible or plausible explanation of the damage. The results were negative.

A metallurgical examination of the spar and fasteners showed evidence of high-cycle, low-stress fatigue along the majority of the upper spar web fractures as well as fatigue cracking in 26 of the 29 fasteners. Only one fastener had failed due to overload. Evaluation of the data indicated that there was no similarity to the damage noted on the accident aircraft; that no single event explains the damage on the United DC-10's upper spar web; and that the damage occurred over a long period of time and was likely to have initiated from

manufacturing discrepancies. Fuselage No. 196 was among those manufactured at the Santa Monica plant where the greatest frequency and number of production discrepancies to the fasteners occurred.

1.17.7 DC-10 Hydraulic and Electrical Systems

Hydraulic power is provided by three hydraulic systems. Each system is powered by two engine-driven hydraulic pumps. Additionally, two electric auxiliary pumps are provided in system No. 3. Emergency hydraulic power is available from one of these auxiliary pumps when powered by the air-driven generator. Two reversible motor-pumps can transfer power from an operating system to an unpressurized system if an engine fails. In addition, two nonreversible motor-pumps can provide a similar transfer of power to certain components of the flight control system.

The three hydraulic systems normally operate independently of each other and are pressurized by their respective engine-driven pumps. The systems power the flight controls, horizontal stabilizer, landing gear, brakes, and nosewheel steering. The two electric auxiliary pumps in hydraulic system No. 3 are primarily for ground use when the engines are shut down; however, auxiliary hydraulic pump No. 1 can be used as an emergency pressure source for the flight controls if all three engines are lost. This can be done inflight by deploying the air-driven generator which will provide electrical power to operate the pump.

The system 1-3 and system 2-3 reversible motor pumps are installed to transfer pressure from an operating hydraulic system to an unpressurized hydraulic system; pressure can be transferred in either direction. No fluid transfer takes place--the transfer of energy is mechanical. Control switches for these pumps are provided on the flight engineer's panel. If the fluid in the reservoir of either the operating system or the system being pressurized falls below a preset minimum, that motor-pump combination will automatically stop operating. Two nonreversible motor pumps are installed in the stabilizer and rudder hydraulic systems to provide backup hydraulic power should the normal power source fail.

Under normal operating conditions, hydraulic power is provided by the two engine-driven pumps in each system. The reversible motor pump controls are in the "arm" position to provide automatic operation in the event of engine failure, and the rudder standby power control switch is in the "arm" position to provide automatic standby power for the rudders if the No. 1 or No. 2 hydraulic system fails.

Except for the spoilers and the upper and lower rudders, each flight control surface is powered by two hydraulic systems. Hydraulic system No. 1 powers the No. 2 and No. 4 spoiler panels on each wing; hydraulic system No. 2 powers the No. 1 and No. 5 spoiler panel on each wing, and hydraulic system No. 3 powers the No. 3 spoiler panel in each wing. The landing gear is powered by the No. 3 hydraulic system.

The lower rudder is powered by hydraulic system No. 2, and its backup power is provided by the 3-2 nonreversible motor pump. The upper rudder is powered by hydraulic system No. 1. Backup power is provided by the 2-1 nonreversible motor pump. Each backup power system has its own independent reservoir and fluid. Consequently, a complete loss of hydraulic fluid in system No. 1 will not affect the operation of the backup system.

The 2-1 nonreversible motor pump also supplies backup power to the horizontal stabilizer, and the operation of the stabilizer trim reduces the fluid flow and pressure available to operate the upper rudder. However, the check valves in the rudder actuator will prevent a drop in hydraulic system pressure from causing a loss of any rudder deflection being held.

When the No. 3 hydraulic system's lines to the outboard slat actuator were severed, during pylon separation, hydraulic fluid began to be lost. The rate of loss was dependent upon the positioning of the slat control valve, and the amount of pinching of the hydraulic lines at the point of severance. According to the chief program engineer for DC-10 design, under the worst case--the control valve wide open and no pinching of the lines--it would require 4 minutes to deplete the reservoir. He further estimated that over a 30-sec to 40-sec period after the rupture there would be no pressure loss and that the retraction of the landing gear would not create significant pressure drain during the time the system remained operable. The witness testified that the hydraulic system was certified in accordance with the existing regulations and compliance with 14 CFR 25.1309 was shown by FMEA and flight testing.

During the early service history of the aircraft, some difficulties with the nonreversible motor pumps were encountered. The pumps were of a new design, and the FMEA's did not predict the in-service difficulties which occurred early in the aircraft's service history. The pumps were redesigned, and the malfunction has not recurred.

According to the witness, there has been only one incident of dual hydraulic system failure. That failure resulted from a tire failure; however, the aircraft was landed safely with one hydraulic system.

The DC-10 electrical system is normally powered by three engine-driven generators. Portions of the system may be powered by a battery and an air-driven generator. The electrical generating system is a.c. with necessary d.c. power provided by transformer rectifier units or a battery. The generators will function either paralleled, unparalleled, or isolated, and each generator can supply enough power to operate all essential electrical systems.

A battery and static inverter combination can provide about 30 min of emergency a.c. and d.c. bus power for the captain's flight instruments, essential communication, and navigation equipment when normal sources are inoperative. The battery and static inverter operations can be obtained by rotating the emergency power switch on the pilot's overhead panel to the "on" position.

Three independent a.c. channels provide power to associated generator buses, which feed associated main a.c. buses. The channels are paralleled through the a.c. tie buses which permits assumption of electrical loads by any functioning generator or generators. The a.c. system is operated normally in parallel with the bus tie relays closed. Two emergency a.c. buses are powered by a.c. buses 1 and 3.

The four transformer rectifier units, which are powered from designated a.c. buses, are the primary sources of d.c. power. Except for transformer rectifier No. 2B, which is powered from the a.c. ground service bus

during ground operation, the other three transformer rectifier systems are similar to their counterpart a.c. systems. However, the d.c. buses are electrically isolated during normal operation.

Protective circuitry automatically isolates faulted buses or components from the other parts of the system. If the protective circuitry senses a generator fault, such as an under voltage condition, the generator relay will open and isolate the generator from its bus; the rest of the system will be powered by the remaining generators. However, if a bus fault is sensed, such as a differential current, the bus tie relay will open and isolate the generator and its associated a.c. buses from the a.c. tie bus. If this occurs, the protective circuitry will also engage a lockout mechanism to protect the remaining buses from damage. The lockout mechanism can be released and power restored to the bus, provided the fault has been cleared by appropriate actions by the flight engineer on his electrical and generator reset panel located at the top of the upper main circuit breaker panel. When he is positioned for takeoff, the flight engineer cannot reach this panel. He must reposition his seat to face his panel, release his safety belt, and get out of his seat to reach the switches. Company procedures only authorize one attempt to restore power. This procedure is not classified as an emergency procedure; it is an "abnormal procedure." The procedure does not contain any immediate action items which must be done without a checklist.

The loss of the No. 1 engine and its associated generator causes a loss of many aircraft systems and instruments. Among these are: The captain's flight instruments, the left stall warning computer, the stickshaker motor, No. 1 engine's instruments, the slat disagree warning light system, portions of the flight control indicating system, portions of the DFDR sensors, and the CVR. In addition to these losses, the flightcrew would be presented with numerous warning lights. The caution and master warning lights on the glareshield would be illuminated. Hydraulic and electrical malfunction lights would be illuminated on the annunciator panel and on the flight engineer's panel. Power to the left a.c. and d.c. emergency buses could have been restored by rotation of the emergency power switch to the "on" position. This action would have powered the left a.c. and d.c. emergency buses and restored the operation of the captain's instruments as well as some of the engine instruments.

1.17.8 Flightcrew Procedures

American Airline's Operating Manual contains the recommended procedures for operating the DC-10 aircraft and its personnel are required to comply with the procedures set forth therein. Since the failure of the pylon and engine did not occur until after V_1, only those company procedures relating to continued flight were examined.[1] These procedures are contained in the Emergency Procedures Section of the Operating Manual.

The Emergency Procedures Section is prefaced with the following guideline:

> "The procedures on the Emergency Checklist are those where immediate and precise action on the part of the crew will substantially reduce the possibility of personal injury or loss of life.

> The emergency procedures in this section are presented as the best way to handle these specific situations. They represent the safest, most practical manner of coping with emergencies, based on the judgment of the most experienced Pilots and F/E's, the FAA approved procedures, and the best available information. If an emergency arises for which these procedures are not adequate or do not apply, the crew's best judgment should prevail."

The manual also provides guidelines as to how the flightcrew will use the emergency checklist. The manual states, in part:

> The checklist is a tool provided to minimize usually hasty and perhaps improper action. Though all checklist procedures are not required to be committed to memory it is expected that all crewmen understand fully each and every procedure.
>
> The nature and seriousness of any given emergency cannot always be immediately and accurately determined. As a professional you will always fly the aircraft and/or immediately correct the obvious prior to any specific reference to the cockpit checklist. Some of the items which fall into the category of attending to the obvious are donning of O_2 masks and goggles, establishing interphone communications, resetting the fire aural warning, etc.

The emergency procedure for a takeoff engine failure, flaps 15° or less or 22°, states, in part:

> "This procedure assumes indication of engine failure where the takeoff is continued. Each takeoff should be planned for the possibility of an engine failure. Normal takeoff procedures ensure the ability to handle an engine failure successfully at any point.
>
> If an engine failure occurs when making a Standard Thrust takeoff, Standard Thrust on the remaining engines will produce the required takeoff performance. If deemed necessary, the remaining engines may be advanced to Maximum Take-Off Thrust.
>
> Speed . . . . . CLIMB OUT AT V_2 UNTIL REACHING 800 FEET AGL OR OBSTACLE CLEARANCE ALTITUDE, WHICHEVER IS HIGHER THEN LOWER NOSE AND ACCELERATE"

The Operating Manual's discussion of the procedure contained an annotated profile drawing of the takeoff. (See figure 14.) The annotations accompanying the profile sketch state (after the aircraft is airborne), "Continue rotation to V_2 (Deck angle 12°-20°)." Over the next picture of the aircraft is the note, "Positive rate-Gear up." The next picture shows the aircraft level at 800 ft AGL and contains the accelerate instructions noted above.

TAKE-OFF ENGINE FAILURE
FLAPS 15° OR LESS OR 22°

This procedure assumes indication of engine failure where the take-off is continued. Each take-off should be planned for the possibility of an engine failure. Normal take-off procedures ensure the ability to handle an engine failure successfully at any point.

If an engine failure occurs when making a Standard Thrust take-off, Standard Thrust on the remaining engines will produce the required take-off performance. If deemed necessary, the remaining engines may be advance to Maximum Take-Off Thrust.

Speed . . . . . CLIMB OUT AT V_2 UNTIL REACHING 800 FEET AFL OR OBSTACLE CLEARANCE ALTITUDE, WHICHEVER IS HIGHER, THEN LOWER NOSE AND ACCELERATE.

At 0°/EXT Min Maneuver Speed,
Flaps . . . . . UP

At V_2 + 50 60 ,
Slats . . . . . .RETRACT
If returning to land, slats may be left extended.

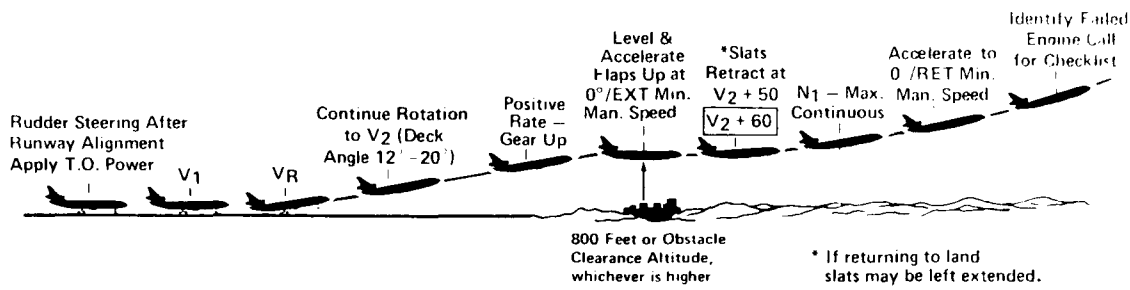

Rudder Steering After Runway Alignment Apply T.O. Power
V1
VR
Continue Rotation to V2 (Deck Angle 12° - 20°)
Positive Rate – Gear Up
Level & Accelerate Flaps Up at 0°/EXT Min. Man. Speed
*Slats Retract at V2 + 50
V2 + 60
N1 – Max. Continuous
Accelerate to 0 /RET Min. Man. Speed
Identify Failed Engine Call for Checklist
800 Feet or Obstacle Clearance Altitude, whichever is higher
* If returning to land slats may be left extended.

Figure 14. Diagram of AAL emergency procedure.

On July 23, 1979, American Airlines issued Operations Bulletin No. DC-10-73 which amended the procedure. The bulletin states, in part:

> "The following climb speeds will be utilized to obstacle clearance altitude when an engine failure occurs after V_1 on takeoff:
>
> - If engine failure occurs after V_1 but not above V_2, maintain V_2 to obstacle clearance altitude.
>
> - If engine failure occurs after V_2, maintain speed attained at time of failure but not above V_2 + 10 to obstacle clearance altitude.
>
> -If engine failure occurs at a speed higher than V_2 + 10, reduce speed to and maintain V_2 + 10 to obstacle clearance altitude.
>
> NOTE:
> If the FD Take-Off mode is engaged at the time of engine failure the Pitch Command Bar (and the Fast/Slow Indicator) will command V_2. Therefore, if the failure occurs above V_2, disregard these indications and fly the speed called for in the above procedure."

1.17.10 Suspension and Restoration of the DC-10 Type Certificate

On June 6, 1979, after a series of postaccident inspections disclosed damaged aft bulkheads in the wing to engine pylons, the Administrator of the FAA issued an Emergency Order of Suspension. The Order suspended the DC-10 series aircraft type certificate "until such time as it can be ascertained that the DC-10 aircraft meets the certification criteria of Part 25 of the FAR and is eligible for a Type Certificate."

On June 26, 1979, the FAA issued Special Federal Aviation Regulation 40 which prohibited the "operation of any Model DC-10 aircraft within the airspace of the United States."

On July 13, 1979, after a series of formal investigations, the Administrator found that the DC-10 met the requirements for issuance of a type certificate. Accordingly, the Emergency Order of Suspension was terminated. (See appendix G.)

2. ANALYSIS

The facts developed during the investigation disclosed that the initial event in the accident sequence was the structural separation of the No. 1 engine and pylon assembly from the aircraft's left wing. Witness accounts, flight data recorder parameters, and the distribution of the major structural elements of the aircraft following the accident provided indisputable evidence that the engine and pylon assembly separated either at or immediately after rotation and about the same time the aircraft became airborne. At that time, the flightcrew was committed to take off, and their decision not to attempt to discontinue takeoff was in accordance with prescribed procedures and was logical and proper in light of information available to them.

The investigation and analysis were concentrated primarily in two major areas. First, the investigation sought to identify the structural failure which led to the engine-pylon separation and to determine its cause; second, the investigation attempted to determine the effects the structural failure had on the aircraft's performance and essential systems, and the operational difficulties which led to the loss of control. In addition, the investigation went beyond these primary areas and probed such areas as the vulnerability of the DC-10's design to maintenance damage, the adequacy of the DC-10's systems to cope with unique emergencies, the quality control exercised during DC-10 manufacturing and aircraft assembly, the adequacy of operator maintenance practices, the adequacy of industry communications of service and maintenance difficulties, the extent of FAA's surveillance of overall industry practices, and the adequacy of an accepted operational procedure.

Pylon Structural Failure

The attachment points of the pylon were examined thoroughly. The fractures and deformations at the separation points in the forward bulkhead and thrust link were all characteristic of overload. The pylon separation began at the aft end in the upper flange of the aft bulkhead, which attached to other elements of the pylon. The upper flange, side flange, and the lower part of the aft bulkhead separated from the remainder of the aft bulkhead and were found on the runway with the engine and pylon structure. The upper portion of the bulkhead containing the spherical bearing remained attached to the wing. Except for the 3 inches of fatigue cracking at the corners of the upper flange, the remainder of the separations and deformations found on the aft bulkhead were all characteristic of overload.

The deformation and fractures at the aft bulkhead's inboard side flange, the thrust attachment, and forward bulkhead indicated that the final separation of the pylon began with a failure at the aft bulkhead which permitted the aft end of the pylon to move down and inboard before total separation. This separation sequence and direction of movement of the pylon before it broke free were consistent with the loads imposed on it during rotation when the combination of aerodynamic loads and thrust imposed a downward vertical tensile load on the bulkhead. The Safety Board could not determine exactly when the aft bulkhead failed, but the weight of the evidence indicated that it most probably failed during the takeoff roll and rotation.

The crescent-shaped deformation on the fracture surface, the shape of which exactly matched the radius of the bottom surface of the wing fitting clevis to which the bulkhead was mated, was strong evidence that the overstress crack in the flange was introduced during removal and installation of the pylon during maintenance. With the bulkhead to clevis attaching hardware in place, the upper surface of the flange was about 0.5 inch below the bottom of the clevis. In order for the clevis to have contacted the flange and deformed the fracture surface, the bolt and bushing through the clevis and the bulkhead's spherical bearing would have had to have been removed. Since this attaching hardware was still in place after the crash, the crescent-shaped deformation was not produced at ground impact and must have been produced when the pylon was installed or removed from the wing.

About 8 weeks before the accident, the No. 1 pylon and engine had been separated from the wing of the accident aircraft in order to replace the spherical bearings in compliance with McDonnell-Douglas' Service Bulletins 54-48 and 54-59. The four other American Airlines and two Continental Airlines aircraft, in which cracks were detected in the aft bulkhead's upper flange, had also been subjected to the same programmed maintenance during which the engine and pylon was removed. Further corroboration that the cracks had been produced during these maintenance operations was obtained when it was learned that Continental Airlines had, on two occasions before the accident, damaged the upper flange on the aft bulkhead as pylons were being removed or reinstalled. In these two instances, the damage was detected; the bulkheads were removed and repaired in accordance with a method approved by McDonnell-Douglas.

Therefore, the evidence indicated that the overstress cracks in the aft bulkhead's upper flange were being introduced during a maintenance operation used by American and Continental Airlines. Both operators had devised special programs to replace the forward and aft bulkhead's spherical bearings. The manufacturer's service bulletins recommended that the maintenance be performed during an engine removal and that the engine be removed from the pylon before the pylon was removed from the wing. Both American Airlines and Continental Airlines believed that it would be more practical to comply with the service bulletin when an aircraft was scheduled for major maintenance--maintenance which would not necessarily otherwise necessitate engine removal. Therefore, American and Continental devised a procedure which they believed to be more efficient than that recommended by McDonnell-Douglas--removal of the engine and pylon as a single unit. An engine stand and cradle were affixed to the engine and the entire weight of the engine and pylon, engine stand, and cradle was supported by a forklift positioned at the proper c.g. for the entire unit. The pylon to wing attaching hardware was removed, and the entire assembly was lowered for access to the spherical bearings. These were replaced and the entire unit was then raised and the attaching hardware reinstalled.

A close examination of these maintenance procedures disclosed numerous possibilities for the upper flange of the aft bulkhead, or more specifically the bolts attaching the spar web to this flange, to be brought into contact with the wing-mounted clevis and a fracture-producing load applied during or after removal of the attaching hardware in the aft bulkhead's fitting. Because of the close fit between the pylon to wing attachments and the minimal clearance between

the structural elements, maintenance personnel had to be extraordinarily cautious while they detached and attached the pylon. A minor mistake by the forklift operator while adjusting the load could easily damage the aft bulkhead and its upper flange. The flange could be damaged in an even more insidious manner; the forks could move imperceptibly as a result of either an internal or external pressure leak within the forklift's hydraulic system during pylon removal. The testimony of the mechanics who performed the maintenance on the accident aircraft confirmed that the procedure was difficult.

Two mechanics stated that they saw the upper lug of the aft bulkhead resting against the bolts attaching the wing-mounted clevis to the wing. To do so would have required a 0.6-inch relative movement between the aft bulkhead and the clevis, relative movement which could only have occurred after the upper flange was deformed. The tests performed by both American Airlines and McDonnell-Douglas following the accident confirmed that a deformation of this magnitude would produce an overload crack.

Except for the 10-inch fracture found on the accident aircraft, the longest maintenance-induced crack found on other upper flanges was 6 inches. Postaccident tests conducted by McDonnell-Douglas and American Airlines indicated that a 6- to 7-inch crack was the longest which could be introduced typically by loading and deforming the flange with a single dynamic impact or steady contact of the flange with the clevis as is believed to have occurred during maintenance.

The accident aircraft's pylon aft bulkhead assembly was the only one in which shims were installed between the bulkhead flange and the attaching spar caps and spar web. The Board believes that the installation of the shims may have had a stiffening effect on the flanges. Load applied to the flange through a spar web attachment bolt by the wing clevis could be spread out through the shims and might have a tendency to produce a longer crack. The shims would also further reduce the clearance between the fastener heads and the lower surface of the wing clevis fitting. Thus, any upward movement of the aft bulkhead would produce a greater downward deflection on a shimmed upper flange than on an unshimmed upper flange. However, the shim might also add strength to the flange and a greater force might be required to crack the shimmed flange. The tests conducted after the accident failed to produce conclusive evidence that installation of shims caused a difference in the damage induced to the flange under similar loading conditions. Thus, the precise effect of the shims remains undetermined.

Tests conducted by McDonnell-Douglas however, did show that repeated load applications could produce a 10-inch crack in the upper flange. This could imply that the upper flange of the accident aircraft contacted the clevis two or more times during the conduct of the maintenance operation. Another possibility proposed by American Airlines which might explain the crack length in the accident aircraft is that the crack occurred in two steps; a crack on the order of 6 inches which occurred during maintenance extended to 10 inches upon the initial application of an abnormal operational load. It was theorized that, in the accident airplane, the installed clearance between the front surface of the aft bulkhead spherical bearing and the rear face of the wing clevis forward ear was less than the nominal minimum clearance of 0.080 in. American Airlines indicated that the aft bulkhead forward flange could have been subjected to a thrust load (tension) of sufficient magnitude to extend the crack during the application of engine takeoff power. To logically explain this possibility it was further theorized

that the tensile load would be transferred to other pylon members thus accounting for stoppage of the crack at the 10-inch length. The investigation could not determine the preaccident tolerances in the aircraft pylon structure; however, other aircraft were found during the postaccident inspections of the DC-10 fleet in which the clevis to bulkhead clearance was sufficiently small that a thrust load would have been imposed on the flange. Further credence is given this theory by the McDonnell-Douglas tests in which it was demonstrated that a flange with a 13-inch preexisting crack including the fatigue growth would not fail unless the vertical and horizontal operating loads were augmented by a thrust load. However, the simulated preexisting damage in this test did not replicate the accident flange and thus the Safety Board did not view this test as conclusive evidence that a thrust load was applied to the bulkhead in the accident aircraft.

While the Safety Board considers the use of shims, the occurrence of repeated flange to clevis impacts, and the application of thrust loads because of improper tolerances as possible factors, other variables such as material grain flow, other material parameters, tolerances, and type of load application might also have resulted in the crack length found in the accident aircraft.

Based upon all the evidence, the Safety Board concludes that the structural separation of the pylon resulted from a complete failure of the forward flange of the aft bulkhead after its residual strength had been critically reduced by a maintenance-induced crack which had been lengthened by service loads.

Aircraft and Flightcrew Performance

The flightcrew of Flight 191 were certificated properly and were qualified for the flight. There was no evidence that their performance was affected by medical problems.

The No. 1 engine and pylon assembly separated after the flightcrew was committed to continuing the takeoff. Witnesses saw the pylon and engine assembly travel up and over the left wing after it separated, and the deformation of the pylon's forward bulkhead was consistent with their observations. The left wing's leading edge skin forward of the pylon's front bulkhead was found on the runway with the pylon structure. There was no evidence that the pylon and engine assembly struck any critical aerodynamic surfaces of the aircraft or any of the flight control surfaces.

Since the loss of thrust provided by the No. 1 engine and the asymmetric drag caused by the leading edge damage would not normally cause loss of control of the aircraft, the Safety Board sought to determine the effects the structural separation had on the aircraft's flight control systems, hydraulic systems, electrical systems, flight instrumentation and warning systems, and the effect, if any, that their disablement had on the pilot's ability to control the aircraft.

As the engine separated from the aircraft, those accessories which were driven by the engine were lost. This included the pumps which provided pressure to the aircraft's No. 1 hydraulic system, and the a.c. generator which provided electrical power to a.c. generator bus No. 1. During a routine emergency wherein the No. 1 engine ceases to operate, all of the services provided by these accessories will remain operable, deriving their respective hydraulic pressure and electrical power from redundant sources driven by one or both of the remaining aircraft engines. However, when the engine separates from the aircraft, the hydraulic pressure and supply lines connecting the pumps with the system are severed, the hydraulic system loses all of its fluid, and thus, hydraulic pressure is not recoverable.

The separation of the engine and pylon also severed the electrical wire bundles inside the pylon. These included the main feeder circuits between the generator and the No. 1 a.c. generator bus. Although this would remove the normal source of power from the bus, the bus could have been powered by the a.c. tie bus, which is powered by generators on the other engines. The No. 1 a.c. generator bus is connected to the a.c. tie bus through a bus tie relay. Protective logic is provided in the aircraft's electrical system. If an electrical fault is detected on the generator bus, the protective logic will cause the bus tie relay to trip, which will open the circuit between the generator bus and the tie bus. This prevents a fault on one generator bus from affecting the aircraft's remaining electrical services. In this accident, the loss of the CVR and certain parameters on the FDR provided evidence that the No. 1 bus tie relay opened when the engine separated, probably as a result of transient short circuits during the separation. The Safety Board concludes that the electrical system's protective circuitry functioned as it was intended and power to the No. 1 generator bus and the services powered by that bus, including d.c. bus No. 1 and left emergency a.c. and d.c. buses, were lost. None of these buses was restored for the remainder of the flight.

The flightcrew might have been able to restore the No. 1 generator bus and all of its services by activating the guarded bus tie relay switch on the electrical and generator reset panel. This action would have been effective only if the bus fault sensed during the separation was temporary. The evidence indicated that the left emergency a.c. and d.c. buses, and the No. 1 d.c. bus could have been restored separately by the activation of the emergency power switch and the No. 1 d.c. tie switch in the cockpit. There was no evidence to indicate that this was done.

The Safety Board believes that the flightcrew probably did not try to restore the lost electrical power, either because of the nature of the overall emergency involving other systems, which they probably perceived to be more critical than the electrical problems, or because the time interval did not permit them to evaluate and respond to the indicated electrical emergency. The Safety Board does not criticize the crew's inaction in this regard; however, since electrical power was not restored, the captain's flight director instrument, several sets of engine instruments and, most importantly, the stall warning and slat disagree warning light systems remained inoperative.

Because of the designed redundancy in the aircraft's hydraulic and electrical systems, the losses of those systems powered by the No. 1 engine should not have affected the crew's ability to control the aircraft. However, as the pylon separated from the aircraft, the forward bulkhead contacted and severed four other hydraulic lines and two cables which were routed through the wing leading edge forward of the bulkhead. These hydraulic lines were the operating lines from the leading edge slat control valve, which was located inboard of the pylon, and the actuating cylinders, which extend and retract the outboard leading edge slats. Two of the lines were connected to the No. 1 hydraulic system and two were connected to the No. 3 system, thus providing the redundancy to cope with a single hydraulic system failure. The cables which were severed provided feedback of the leading edge slat position so that the control valve would be nulled when slat position agreed with position commanded by the cockpit control.

The severing of the hydraulic lines in the leading edge of the left wing could have resulted in the eventual loss of No. 3 hydraulic system because of fluid depletion. However, even at the most rapid rate of leakage possible, the system would have operated throughout the flight. The extended No. 3 spoiler panel on the right wing, which was operated by the No. 3 hydraulic system, confirmed that this hydraulic system was operating. Since two of the three hydraulic systems were operative, the Safety Board concludes that, except for the No. 2 and No. 4 spoiler panels on both wings which were powered by the No. 1 hydraulic systems, all flight controls were operating. Therefore, except for the significant effect that the severing of the No. 3 hydraulic system's lines had on the left leading edge slat system, the fluid leak did not play a role in the accident.

During takeoff, as with any normal takeoff, the leading edge slats were extended to provide increased aerodynamic lift on the wings. When the slats are extended and the control valve is nulled, hydraulic fluid is trapped in the actuating cylinder and operating lines. The incompressibility of this fluid reacts against any external air loads and holds the slats extended. This is the only lock provided by the design. Thus, when the lines were severed and the trapped hydraulic fluid was lost, air loads forced the left outboard slats to retract. While other failures were not critical, the uncommanded movement of these leading edge slats had a profound effect on the aerodynamic performance and controllability of the aircraft. With the left outboard slats retracted and all others extended, the lift of the left wing was reduced and the airspeed at which that wing would stall was increased. The simulator tests showed that even with the loss of the No. 2 and No. 4 spoilers, sufficient lateral control was available from the ailerons and other spoilers to offset the asymmetric lift caused by left slat retraction at airspeeds above that at which the wing would stall. However, the stall speed for the left wing increased to 159 KIAS.

The evidence was conclusive that the aircraft was being flown in accordance with the carrier's prescribed engine failure procedures. The consistent 14° pitch attitude indicated that the flight director command bars were being used for pitch attitude guidance and, since the captain's flight director was inoperative, confirmed the fact that the first officer was flying the aircraft. Since the wing and engine cannot be seen from the cockpit and the slat position indicating system was inoperative, there would have been no indication to the flightcrew of the slat retraction and its subsequent performance penalty. Therefore, the first officer continued to comply with carrier procedures and maintained the

commanded pitch attitude; the flight director command bars dictated pitch attitudes which decelerated the aircraft toward V_2, and at V_2 + 6, 159 KIAS, the roll to the left began.

The aircraft configuration was such that there was little or no warning of the stall onset. The inboard slats were extended, and therefore, the flow separation from the stall would be limited to the outboard segment of the left wing and would not be felt by the left horizontal stabilizer. There would be little or no buffet. The DFDR also indicated that there was some turbulence, which could have masked any aerodynamic buffeting. Since the roll to the left began at V_2 + 6 and since the pilots were aware that V_2 was well above the aircraft's stall speed, they probably did not suspect that the roll to the left indicated a stall. In fact, the roll probably confused them, especially since the stickshaker had not activated.

The roll to the left was followed by a rapid change of heading, indicating that the aircraft had begun to yaw to the left. The left yaw -- which began at a 4° left wing down roll and at 159 KIAS--continued until impact. The abruptness of the roll and yaw indicated that lateral and directional control was lost almost simultaneous with the onset of the stall on the outboard section of the left wing.

The simulator tests showed that the aircraft could have been flown successfully at speeds above 159 KIAS, or if the roll onset was recognized as a stall, the nose could have been lowered, and the aircraft accelerated out of the stall regime. However, the stall warning system, which provided a warning based on the 159 KIAS stall speed, was functioning on the successful simulator flights. Although several pilots were able to recover control of the aircraft after the roll began, these pilots were all aware of the circumstances of the accident. All participating pilots agreed that based upon the accident circumstances and the lack of available warning systems, it was not reasonable to expect the pilots of Flight 191 either to have recognized the beginning of the roll as a stall or to recover from the roll. The Safety Board concurs.

In addition, the simulator tests showed that the aircraft could have been landed safely in its accident configuration using then current American Airlines procedures. The simulator tests also disclosed that the aircraft could have been landed with an asymmetric leading edge slat configuration. The speed margins during the final positions of the landing approach are also very small; however, the landing situation is considered less critical since additional thrust is readily available as required to either adjust the flightpath or accelerate the aircraft. In addition, service experience has shown that loss of slats on one wing during the approach presents no significant control problems.

The pilot's adherence to the airspeed schedules contained in the company's engine-out emergency procedure resulted in the aircraft's entering the stall speed regime of flight. Had the pilot maintained excess airspeed, or even V_2 + 10, the accident may not have occurred. Since the airspeed schedules contained in American Airlines' emergency procedures at the time of the accident were identical to those currently contained in the emergency procedures of other air carriers, the Safety Board believes that speed schedules for engine-out climb profiles should be examined to insure that they afford the maximum possible protection.

In summary, the loss of control of the aircraft was caused by the combination of three events: the retraction of the left wing's outboard leading edge slats; the loss of the slat disagreement warning system; and the loss of the stall warning system -- all resulting from the separation of the engine pylon assembly. Each by itself would not have caused a qualified flightcrew to lose control of its aircraft, but together during a critical portion of flight, they created a situation which afforded the flightcrew an inadequate opportunity to recognize and prevent the ensuing stall of the aircraft.

DC-10 Design and Certification

The pylon design, and in particular the aft bulkhead and its upper flange, satisfied the fail-safe requirements of the 1965 Federal Aviation Regulations. The stress analysis of the pylon structure showed that the stress level in the upper flange of the aft bulkhead was well below the fatigue damage level and the material was not considered to be vulnerable to stress corrosion. Therefore, since it was not necessary to apply fail-safe criteria to the flange, the design did not provide an alternate path for the transmittal of loads in the event the flange failed. Although the flight tests conducted after the accident disclosed that additional thrust loads were being imposed on the aft bulkhead which were not accounted for in the original certification analysis, the stress levels were still below the fatigue-damage level. In addition, postaccident tests and analyses of alternate load paths for other pylon structural members showed that, even with a failed thrust link, the bulkheads could carry the takeoff thrust load. Furthermore, the postaccident inspections of the DC-10's did not disclose any evidence of fatigue damage on any of the bulkheads within the fleet. Therefore, the Safety Board finds that the original certification's fatigue-damage assessment of fatigue damage was in conformance with the existing requirements.

The Damage-Tolerance concept embodied in the December 1, 1978, amendment to 14 CFR 25.571 levies different requirements on the certification of structural design. While the regulations in effect prior to the adoption of this amendment considered susceptibility of undamaged structure to fatigue, this new concept requires that an evaluation of the strength, detail design, and fabrication must show that catastrophic failure due to fatigue, corrosion, or accidental damage will be avoided throughout the operational life of the aircraft. The evaluation must include a determination of the probable locations and modes of damage due to fatigue, corrosion, or accidental damage. If a part is determined to be susceptible to these types of damage, its operational life must be established by analysis and supporting tests. The operational life must be consistent with the onset of damage and its subsequent growth during testing. The results of these tests and analyses are used to establish inspection areas and frequencies to monitor the structural integrity of the part.

Had the requirement for accidental damage evaluation been in effect when the the DC-10 was designed, one might expect that such consideration would have been given to accidental damage to the upper flange of the pylon aft bulkhead. However, this would still have depended upon the interpretation of the type of accidental damage required to be considered. The manufacturer contends that accidental damage should be limited to damage which can be inflicted during routine aircraft maintenance or servicing, such as contact at galley and cargo

doors or dropping of tools in areas of frequent maintenance. Based on this interpretation, the accidental contact between the pylon aft bulkhead and the wing-mounted clevis probably would not have been considered since it did not constitute routine maintenance. And, even had this accidental contact been considered, the design may not have been different; however, more stringent inspection requirements might have been imposed, particularly following maintenance. Following the accident, the FAA required McDonnell-Douglas to conduct a damage-tolerance assessment of the pylon structure in accordance with the new regulation. When the program was conducted it was presumed that a crack in the bulkhead flange could be detected visually before it was 3 inches long and that the residual strength of the damaged element would far exceed the operational load requirements. Based on these criteria, the analysis and tests showed that the design meets the current damage-tolerance requirement.

Although the design of the pylon complied with the strength requirements of the regulations, the Safety Board believes that neither the designers nor the FAA certification review team adequately considered the vulnerability of the structure to damage during maintenance. In several places, clearances were unnecessarily small and made maintenance difficult to perform. Historically, pylons have had to be lowered and replaced for many reasons, such as ground accidents, fatigue, and corrosion. In fact, parts of the pylon structure are either on a sampling inspection or 100-percent inspection schedule. Under these circumstances, McDonnell-Douglas should have foreseen that pylons would be removed, and therefore, the mating parts of the aft bulkhead should have been designed to eliminate, or at least minimize, vulnerability to damage during maintenance. Whenever major components are made up of parts that can be removed, the design must protect each part from damage during removal or reinstallation. Either the parts should be made strong enough to withstand inadvertent contact, or clearances should be provided that will not allow contact. The pylon aft bulkhead could have been designed so that the upper part of the lug would bottom on the base of the wing-mounted clevis, before the upper spar web and aft bulkhead flange assembly contacted the clevis ear. On the actual design there is only .080-inch clearance between the bolt heads on the flange assembly and the clevis with the pylon installed. With adverse tolerances, this clearance of the fitting can be reduced to less than .030 inch. The evidence, provided by a dimensional analysis, which included the thickness of the shims, showed that an interference fit of about .030 inch could have existed. Following the accident, interference was also found in some other aircraft in which shims were installed.

In order to reinstall a pylon with an interference fit between the aft bulkhead flange assembly and the wing clevis, the flange assembly would have to be brought into contact with the wing clevis and the flange would have to be loaded and deflected enough to allow the bushing and bolt to be inserted through the clevis and spherical joint. Although tests showed that the load required to create this deflection would not fracture the flange, the maintenance operation, regardless of the procedures used, would be difficult to perform and would be particularly vulnerable to damage-producing errors. Thus, the Safety Board concludes that the basic design of the aft attachment of the pylon to the wing was unnecessarily vulnerable to maintenance damage.

The Safety Board is also concerned that the designs of the flight control, hydraulic, and electrical systems in the DC-10 aircraft were such that all

were affected by the pylon separation to the extent that the crew was unable to ascertain the measures needed to maintain control of the aircraft.

The airworthiness regulations in effect when the DC-10 was certificated were augmented by a Special Condition, the provisions of which had to be met before the aircraft's fully powered control system would be certificated. The Special Condition required that the aircraft be capable of continued flight and of being landed safely after failure of the flight control system, including lift devices. These capabilities must be demonstrated by analysis or tests, or both. However, the Special Condition, as it applied to the slat control system, was consistent with the basic airworthiness regulations in effect at the time. The basic airworthiness regulations specified requirements for wing flap asymmetry only and did not include specific consideration of other lift devices. Because the leading edge slat design did not contain any novel or unusual features, it was certificated under the basic regulation. The flap control requirements for symmetry and synchronization were applied to and satisfied by the slat system design. Since a malfunction of the slat actuating system could disrupt the operation of an outboard slat segment, a fault analysis was conducted to explore the probability and effects of both an uncommanded movement of the outboard slats and the failure of the outboard slats to respond to a commanded movement. The fault analysis concluded that the aircraft could be flown safely with this asymmetry.

Other aircraft designs include positive mechanical locking devices to prevent movement of slats by external loads following a primary failure. The DC-10 design did not include such a feature nor was it deemed necessary, since compliance with the regulations was based upon analysis of those failure modes which could result in asymmetrical positioning of the leading edge devices and a demonstration that sufficient lateral control was available to compensate for the asymmetrical conditions throughout the aircraft's flight envelope. The flight tests conducted to evaluate the controllability of the aircraft were limited to a minimum airspeed compatible with stall-warning activation predicated upon the slat-retracted configuration.

The takeoff regime at lower airspeeds was not examined in flight. However, analysis of the takeoff regime showed that, with all engines operating, the aircraft would be accelerated to and maintain a positive stall margin throughout the flight. The analysis also showed that if a loss of engine thrust and slat retraction were to occur during takeoff, the aircraft's capability to accelerate to and maintain a positive stall margin was compromised. Further consideration of this hazardous combination was limited to a mathematical probability projection, which showed that the combination was extremely improbable. Thus, the design was accepted as complying with the requirements. If the structural loss of a pylon had been included in the probability projection, the vulnerability of the hydraulic lines and position feedback cables may have influenced adversely the probability projection.

Also, the influence on aircraft control of the combined failure of the hydraulic and electrical systems was not considered. When aircraft controllability was first evaluated based on asymmetric leading edge devices, it was presumed that other flight controls would be operable and that slat disagree and stall warning devices would be functioning. Flight 191 had accelerated to an airspeed at which an ample stall margin existed. Postaccident simulator tests showed that, if the

airspeed had been maintained, control could have been retained regardless of the multiple failures of the slat control, or loss of the engine and Nos. 1 and 3 hydraulic systems. On this basis alone, the Safety Board would view the design of the leading edge slat system as satisfactory. However, the additional loss of those systems designed to alert the pilot to the need to maintain airspeed was most critical. The stall warning system lacked redundancy; there was only one stickshaker motor; and the left and right stall warning computers did not receive crossover information from the applicable slat position sensors on opposite sides of the aircraft. The accident aircraft's stall warning system failed to operate because d.c. power was not available to the stickshaker motor. Even had d.c. power been available to the stickshaker motor, the system would not have provided a warning based on the slats retracted stall speed schedule, because the computer receiving position information from the left outboard slat was inoperative due to the loss of power on the No. 1 generator bus. Had power been restored to that bus, the system would have provided a warning based on the slat retracted stall speed. However, in view of the critical nature of the stall warning system, additional redundancy should have been provided in the design.

In summary, the certification of the DC-10 was carried out in accordance with the rules in effect at the time. The premises applied to satisfy the rules were in accordance with then accepted engineering and aeronautical knowledge and standards. However, in retrospect, the regulations may have been inadequate in that they did not require the manufacturer to account for multiple malfunctions resulting from a single failure, even though that failure was considered to be extremely improbable. McDonnell-Douglas considered the structural failure of the pylon and engine to be of the same magnitude as a structural failure of a horizontal stabilizer or a wing. It was an unaceptable occurrence, and therefore, like the wing and horizontal stabilizer, the pylon structure was designed to meet and exceed all the foreseeable loads for the life of the aircraft. Therefore, just as it did not analyze the effect the loss of a wing or horizontal stabilizer would have on the aircraft's systems, McDonnell-Douglas did not perform an analysis based on the loss of the pylon and engine.

Logic supports the decision not to analyze the loss of the wing and horizontal stabilizer. With the loss of either of these structures, further flight is aerodynamically impossible and the subsequent effect of the loss on the aircraft's systems is academic. However, similar logic fails to support the decision not to analyze the structural failure and loss of the engine and pylon, since the aircraft would be aerodynamically capable of continued flight. The possiblity of pylon failure, while remote, was not impossible. Pylons had failed. Therefore, fault analyses should have been conducted to consider the possible trajectories of the failed pylon, the possibilities of damage to aircraft structure, and the effects on the pilot's ability to maintain controlled flight. Since the capability of continued flight was highly probable, the fault analysis might have indicated additional steps or methods which could have been taken to protect those systems essential to continued flight.

Therefore, the Safety Board concludes that the design and interrelationship of the essential systems as they were affected by the structural loss of the pylon contributed to this accident.

Manufacturing and Quality Control

The Safety Board did not determine whether the installation of the shims in the pylon aft bulkhead and spar web assembly was a factor in this accident. However, the inclusion of these shims on certain aircraft raised concerns regarding the adequacy of manufacturing quality control. These concerns were heightened when several pylons were found to have failed, loose, or missing fasteners, including the significantly damaged pylon on the United Airlines DC-10, N1827U. These too were attributable to production deficiencies at the McDonnell-Douglas facilities where the pylons were assembled.

Beginning with fuselage No. 15 and ending with fuselage No. 36, there were 23 pylons which required the insertion of 10-inch-long, .050-inch-thick shims. Within these 21 fuselages, there was interference between the bottom of the clevis and the fastener heads on 7 pylons.

A Rejection and Disposition Item had been issued for the shim on the accident aircraft; however, the evidence was inconclusive as to whether the proper procedures were followed in issuing the authorization. The authorization was signed by a liaison engineer, indicating that a stress analysis had been made on the effects of the shim. However, there was no evidence that a dimensional analysis had been conducted to determine whether the insertion of the shims would adversely affect clearance during assembly of the pylon to the wing, a clearance which could affect the loads imposed on primary structural elements. The clearance problem on the upper flange was resolved by repositioning locator pins on the tooling jigs; however, before this solution was found, 21 fuselages had passed through the line and 23 pylons had required the shim. The Safety Board believes that proper quality control procedures would have brought about an earlier resolution of this problem.

In addition, despite inspection and quality control procedures, 31 fuselages left the assembly line with defective pylon fasteners. The number of defective fasteners ranged from a minimum of 2 to a maximum of 26, found on the United Airlines DC-10.

In summary, the evidence showed that there were deficiencies in the pylon assembly line procedures at McDonnell-Douglas and that the quality-control procedures in effect did not detect and effect a timely correction of these deficiencies. While these were not causal to this accident, the Safety Board believes that they illustrate deficiencies of the type which could lead to accidents.

Maintenance Programs

Although the Safety Board believes that the design of the pylon structure was less than optimum with regard to maintainability, the evidence is conclusive that many pylons were removed from the wing and reinstalled without imposing damage to the structure. There is no doubt, however, that this maintenance operation requires caution and extreme precision because of the minimal clearances at the pylon-to-wing attachment points and the danger of inadvertent impact of the structure.

McDonnell-Douglas was apparently aware of the precision which would be required, and as a result it specified in its original maintenance procedures and subsequent service bulletins that the engine be separated from the pylon before the pylon is removed from the wing. While removal of the engine would not completely eliminate the possibility of imposing damage to the pylon structure, the likelihood would certainly be much less than that which existed when handling the pylon and engine as single unit. The pylon assembly without the engine weighs about 1,865 lbs and the c.g. is located approximately 3 ft forward of the forward bulkhead attachment points. The pylon and engine together weigh about 13,477 lbs. and the c.g. is located about 9 ft forward of the forward bulkhead attachment points. With the engine removed, the pylon can be supported relatively close to the pylon-to-wing attachment points where precise relative motion between the pylon and wing structure can be closely observed and controlled. Thus, McDonnell-Douglas did not encourage removing the engine and pylon assembly as a single unit because of the risk involved in remating the combined assembly to the wing attach points. The Safety Board, therefore, is concerned with the manner in which the procedures used to comply with Service Bulletins 54-48 and 54-59 were evaluated, established, and carried out.

American Airlines is a designated alteration station, as are the other major carriers that conduct heavy maintenance programs. Pursuant to that designation and the applicable regulations, carriers are authorized to conduct major maintenance in accordance with the maintenance and inspection program established by the FAA's Maintenance Review Board when the aircraft was introduced into service. Carriers are also authorized to conduct alterations and repairs in accordance with the procedures set forth in its maintenance manuals or established by its engineering departments. The FAA, through its principal maintenance inspectors, is responsible for surveillance of carriers' maintenance programs. However, this surveillance is broadly directed toward insuring that the carriers comply with the established maintenance and inspection program and that their maintenance programs, including administration, general practices, and personnel qualifications, are consistent with practices acceptable to the Administrator. The FAA can review the carriers' maintenance manual, but its formal approval is not required. Carriers are permitted to develop their own step-by-step maintenance procedures for a specific task without obtaining the approval of either the manufacturer of the aircraft or the FAA. It is not unusual for a carrier to develop procedures which deviate from those specified by the manufacturer if its engineering and maintenance personnel believe that the task can be accomplished more efficiently by using an alternate method.

Thus, in what they perceived to be in the interest of efficiency, safety, and economy, three major carriers developed procedures to comply with the changes required in Service Bulletins 54-48 and 54-59 by removing the engine and pylon assembly as a single unit. One carrier apparently developed an alternate procedure which was used without incident. However, both American Airlines and Continental Airlines employed a procedure which damaged a critical structural member of the aircraft. The procedure, developed by American Airlines and issued under ECO R-2693, was within American Airlines' authority, and approval or review was neither sought nor required from the manufacturer or the FAA.

The evidence indicated that American Airlines' engineering and maintenance personnel implemented the procedure without a thorough evaluation to insure that it could be conducted without difficulty and without the risk of damaging the pylon structure. The Safety Board believes that a close examination of the procedure might have disclosed difficulties that would have concerned the engineering staff. In order to remove the load from the forward and aft bulkhead's spherical joints simultaneously, the lifting forks had to be placed precisely to insure that the load distribution on each fork was such that the resultant forklift load was exactly beneath the c.g. of the engine and pylon assembly. To accomplish this, the forklift operator had to control the horizontal, vertical, and tilt movements with extreme precision. The failure of the ECO to emphasize the precision this operation required indicates that engineering personnel did not consider either the degree of difficulty involved or the consequences of placing the lift improperly. Forklift operators apparently did not receive instruction on the necessity for precision, and the maintenance and engineering staff apparently did not conduct an adequate evaluation of the forklift to ascertain that it was capable of providing the required precision.

The evidence showed that during the actual maintenance, the forklift operator used the supported weight gauge to adjust the forklift; however, the adjustment was made in a trial-and-error fashion until the attaching hardware was removed from the forward bulkhead. If the load applied by the forklift with respect to the c.g. of the assembly was not balanced, a load would be applied at the aft bulkhead attachment joint. Thus, after the maintenance personnel removed the forward bulkhead and thrust link attachments, they would not be able to remove the loaded bolt and bushing from the aft bulkhead fitting until the forklift was repositioned. More precision was required to reposition the forklift because of the 15-ft distance from the forklift to the aft bulkhead. If the bolt and bushing were forced out of the bulkhead attachment while under load, the aft end of the pylon could move and the upper spar web to aft bulkhead flange attachment bolts could strike the forward lug of the wing clevis and apply a bending load to the flange. Whether the force applied through the contacting surfaces would be enough to damage the pylon structure depended upon the alignment precision of the forklift operator.

Maintenance personnel testified that it was difficult to adhere to the removal sequence of the attaching hardware that was specified in the ECO. The Safety Board believes that the difficulty encountered in repositioning the forklift to remove the load at the aft bulkhead was the basis for such an assessment. As a result, maintenance personnel altered the sequence of hardware removal, and removed the attachments at the aft bulkhead before those of the forward bulkhead. Using this procedure, the forklift was positioned to remove the load at the aft bulkhead first. Although this still required extreme precision, the pivot action at the attached forward bulkhead reduced the lever arm over which minor misalignments of the forklift would act. However, while easing the task of removing the aft bulkhead fitting, the change to this sequence greatly increased the risk of damage to the pylon structure.

After the bolt and bushing were removed from the aft bulkhead attachment joint, the forward bulkhead would continue to act as a pivot. Thus, any advertent or inadvertent vertical movement of the forklift would result in a vertical movement at the pylon's aft bulkhead. If the lifting forks were lowered, the spar web attachment at the aft bulkhead flange would be brought into contact with the forward lug of the wing clevis; and if the forks were lowered further, the supported weight of the combined assembly would be transferred from the forklift to the aft bulkhead to wing clevis contact. As the load is transferred, it will increase and can eventually reach the limit wherein the aft bulkhead is reacting the total moment about the forward bulkhead created by the weight of the combined assembly and the 9 ft distance to the c.g. This could impose a load of over 20,000 lbs. on the aft bulkhead (which is about 6 ft from the forward bulkhead). The tests showed that the flange would fracture with a load application of less than 8,000 lbs. 7/

Had a proper evaluation of this procedure been conducted, it should have been apparent that there are two probable reasons which would cause lifting forks to lower: First, it was almost certain that the forklift operator would have to readjust the forklift to relieve the load in the forward bulkhead spherical joints before their removal. In doing so, it was conceivable that he would operate both elevation and tilt controls in a manner which would momentarily lower the lifting forks. Second, any removal of power from the forklift when combined with external or internal leakage within the forklift hydraulic system would result in a slow descent of the lifting forks.

The testimony at the public hearing disclosed that the forklift had to be repositioned after the attachment hardware of the aft bulkhead was removed during a pylon removal, and the evidence indicated that this occurred on the accident aircraft. The testimony also indicated that the forklift was not powered for a period of time because it ran out of fuel. The postaccident forklift tests showed that, under these conditions, leakage would allow a drift down of 1 inch in 30 min. The postaccident flange loading tests showed that a movement of 0.4 inch or less at the c.g. would produce a 7-inch fracture of the flange.

The evidence also showed that, in two instances at Continental Airlines, the sound caused by the flange fracturing was heard by maintenance personnel. The fact that it was not heard by maintenance personnel in the other cases can probably be attributed to several factors: the surrounding noise level in the work area; the locations of the maintenance personnel when the flange broke; the sound produced by the fracture may not have been as loud; or a combination of all these factors. The Safety Board, therefore, concludes that there were several possibilities wherein the aft bulkhead flange could have been damaged, and that such damage could have occurred without being detected by maintenance personnel working in the vicinity of the pylon. The Safety Board also concludes that these hazards might have been detected had a proper evaluation been conducted.

7/ With the engine removed, the maximum load which can be imposed on the aft bulkhead because of the moment about the forward bulkhead is about 900 lbs.

There was no evidence that the American Airlines maintenance personnel informed either the engineering or quality control department about the difficulties they encountered using the ECO sequence or that the sequence was changed. Had they done so, it was possible that the cognizant engineers would have examined more closely the entire operation from the standpoint of damage risk. However, the testimony at the public hearing indicated that without specific instructions to the contrary, sequential adherence to the times in the ECO was not mandatory. Also, there was no evidence to show that either engineering or quality control personnel routinely examine maintenance procedures in a step-by-step fashion to determine whether such procedures were particularly damage-inducing.

The Safety Board believes that other shortcomings were evident in the ECO and in the manner in which it was implemented on the accident aircraft. The ECO did not specify in the requirement for inspection that the pylon structure be inspected either before or after it was reinstalled on the wing. The evidence showed that the pylon upper spar web to the mating structure attachment was inspected and the new sealant was installed after the assembly was lowered from the wing. However, the Safety Board could not determine the extent to which the aft bulkhead flange was examined. Furthermore, the ECO did not require that quality control personnel inspect either the pylon structure or the pylon to wing attachment hardware after the pylon was reinstalled to the wing. While this omission does not appear to be a factor in this accident, it does present the potential for an accident-producing error to escape detection.

The Safety Board, therefore, concludes that there were other deficiencies within the American Airlines maintenance program, some of which contributed to this accident. Among these was the failure of engineering department to ascertain the damage-inducing potential of a procedure which deviated from the manufacturer's recommended procedure, their failure to adequately evaluate the performance and condition of the forklift to assure its capability for the task, the absence of communications between maintenance personnel and engineers regarding difficulties encountered and the procedural changes which were required in the performance of the pylon maintenance, and the failure to establish an adequate inspection program to detect maintenance-imposed damage. Although the Safety Board directed its investigation to American Airlines, the Safety Board is concerned that these shortcoming were not unique to that carrier. Since two of Continental Airlines DC-10's were found to have been flying with damaged bulkheads, similar shortcomings were also present in its maintenance program.

The Safety Board is also concerned about broader issues of maintenance and inspection as they relate to the program established by the DC-10 Maintenance Review Board when the aircraft was initially introduced into service. While the inspection program appears to be monitoring the on-condition maintenance process adequately, the postaccident investigations of the DC-10 fleet disclosed areas where shortcomings within the inspection program may exist.

Much of the DC-10's pylon structure was subject to a sampling inspection program. Consequently, damage related to manufacturing deficiencies on certain aircraft would only have been detected if one of those aircraft had been among the population of the inspected sample. Therefore, the failures of the

fasteners and the cracks on the upper spar web of United Airlines DC-10 N1827U would not have been detected had it not been for the inspections required as a result of the accident. The area in which the damage was located was not a 100-percent inspection area. While other items on the pylon near the damaged area were under the 100-percent inspection program and the inspector should, when the area was opened for the required inspection, inspect all areas that are visible, the probability of finding the damage would be limited by the area open to his vision and the manner in which he conducted the inspection. Despite the requirements for the 100-percent inspections on nearby pylon structure, the probability of detecting damage in an adjacent area was not good, as evidenced by the 31 aircraft with loose, failed, or missing fasteners discovered during postaccident inspections. The facts indicate that inspection requirements should be established that will allow for the detection of these types of discrepancies.

The Safety Board is also concerned that, as indicated in this accident, significant structural items can be damaged during major maintenance without the knowledge of the personnel performing the task. The postaccident investigation disclosed that six DC-10's were returned to service with a cracked flange in the pylon aft bulkhead, one of which contained fatigue cracks at the ends of the fracture. The evidence points out the necessity for establishing inspection programs that will insure that significant structural items damaged in this manner can be detected before they are returned to service and to reassure, by a later inspection, that the repaired structure and the structure which has been exposed to major maintenance has not been damaged or flawed.

In summary, the Safety Board believes that the criteria used by the Maintenance Review Board to establish inspection requirements should be reviewed to determine their adequacy for insuring detection of damage to structurally significant parts which can result from faults introduced during manufacturing, assembly, and maintenance operations.

Industry Communications Regarding Maintenance Difficulties

The Safety Board is particularly concerned that because of the limitations of the current reporting system the FAA and key engineering and maintenance personnel at American Airlines were not aware that Continental Airlines had damaged two aft bulkhead flanges on two of its DC-10's until after the accident. In December 1978, after it discovered the first damaged bulkhead, Continental apparently conducted a cursory investigation and determined that the damage resulted from a maintenance error. A repair was designed for the bulkhead and was submitted to McDonnell-Douglas for stress analysis approval. The repair was approved and performed, and the aircraft returned to service.

On January 5, 1979, Operational Occurrence Report No. 10-7901 was published by McDonnell-Douglas. The publication contained descriptions of several DC-10 occurrences involving various aircraft systems, personnel injury, and the damage inflicted on the Continental Airlines DC-10. The report described the damage to the upper flange of the Continental aircraft and indicated that it occurred during maintenance procedures used at the time it was damaged. However, the way in which the damage was inflicted was not mentioned. The manufacturer had no authority to investigate air carrier maintenance practices and, therefore, accepted the carrier's evaluation of how the flange was damaged.

Since the damage was inflicted during maintenance, 14 CFR 21.3 relieved McDonnell-Douglas of any responsibility to report the mishap to the FAA. Although American Airlines was on the distribution list for Operational Occurrence Reports, testimony disclosed that the maintenance and engineering personnel responsible for the pylon maintenance were not aware of the report.

Continental Airlines discovered the damage to the second bulkhead in February 1979. Again the carrier evaluation indicated that the cause of the damage was related to personnel error, and that there was apparently no extensive effort to evaluate the engine-pylon assembly removal and reinstallation procedures. The bulkhead was also repaired using the procedure previously approved by McDonnell-Douglas.

The carrier did not report the repairs that were made to the two bulkheads to return them to service, and there was no regulatory requirement to do so. What constitutes a major repair may be subject to interpretation, but what is to be reported is not. The bulkheads were not altered; they were repaired. Even had the repairs been classified by the carrier as major, 14 CFR 121.707(b) only requires that a report be prepared and kept available for inspection by a representative of the FAA. Second, the regulation does not indicate that the contents of the required report include a description of the manner in which the damage was inflicted. The regulation and the evidence indicated that the purpose of the reports was to permit the FAA to evaluate the end-product to insure that the basic design of the repaired or altered part had not been changed.

The Mechanical Reliability Reporting criteria of 14 CFR 121.703 requires the certificate holder to report "the occurrence or detection of each failure, malfunction, or defect concerning. . ." and then lists 16 criteria to which these apply. The FAA and apparently the aviation industry have traditionally interpreted 121.703 to apply to only service-related problems, which would therefore exclude reporting of the flange damage caused by maintenance. In view of this interpretation, the Board concludes that there is a serious deficiency in the reporting requirements which should be corrected.

Therefore, the Safety Board concludes that neither the air carrier nor the manufacturer interpreted the regulation to require further investigation of the damages or to report the damage to the FAA. However, the Safety Board views the omission of such requirements as a serious deficiency in the regulations.

The Safety Board also believes that regardless of the regulation, Continental Airlines had the opportunity and should have conducted a thorough investigation into the damage risk involved in the procedure being used to accomplish the pylon maintenance, particularly after it was known that two bulkheads had been damaged. Certainly, the possibility that similar damage could have occurred on other aircraft without detection and the possibility that other carriers using similar maintenance procedures could encounter the same problem should have been considered. Had a more thorough investigation been conducted, the incident might have been given more emphasis in the report to the other carriers. Action then might have been taken to revise the maintenance procedure and to inspect those aircraft which had been exposed to the potential damage.

McDonnell-Douglas did not investigate Continental Airlines' maintenance procedures and accepted its finding that the damage was due to maintenance error. However, 2 months later McDonnell-Douglas received the report that a second bulkhead was damaged, that the location and type of damage was almost identical to the damage inflicted on the first bulkhead, and that the damage was again due to maintenance error. McDonnell-Douglas then had the opportunity to question whether maintenance error was the result of a procedural problem rather than accepting personnel error as the cause. They should have investigated the procedure and perhaps discovered the flaws within the procedure. However, they accepted the company's evaluation of cause and did not pursue the matter further.

The Safety Board, therefore, believes that the regulatory reporting structure had and still has a serious deficiency. Damage to a component identified as "structurally significant" must be reported in a manner which will assure that the damage and the manner in which it is inflicted is evaluated, and the results of that evaluation disseminated to the operators and airframe manufacturers. Second, damage to a component of this type should be reported regardless of whether it was incurred during flight, ground operations, or maintenance. Finally, damage suffered by these types of structures should be investigated by representatives of the operator, airframe manufacturer, and the Administrator.

Surveillance of Industry Practices by Federal Aviation Administration

The Safety Board believes that the facts, conditions, and circumstances of this accident and the information obtained during the investigation illustrate deficiencies in the aviation industry ranging from aircraft design through operations. The Safety Board recognizes that resource limitations prohibit the FAA from exercising rigid oversight of all facets of the industry. Therefore, the FAA must exercise its authority by insuring that aircraft designs do comply with regulations, that manufacturers quality control programs are effective, that aircraft operators adhere to a proper maintenance program; and that operational procedures adopted by the carriers consider even unique emergencies which might be encountered.

In summary, the Safety Board recognizes that the overall safety record of the current generation of jet aircraft clearly indicates that the regulatory structure under which U.S. commercial aviation operates and the industry's commitment to safety is basically sound. The Safety Board, however, is concerned that this accident may be indicative of a climate of complacency. Although the accident in Chicago on May 25 involved only one manufacturer and one carrier, the Safety Board is concerned that the nature of the identified deficiencies in design, manufacturing, quality control, maintenance and operations may reflect an environment which could involve the safe operation of other aircraft by other carriers.

3. CONCLUSIONS

3.1 Findings

1. The engine and pylon assembly separated either at or immediately after liftoff. The flightcrew was committed to continue the takeoff.

2. The aft end of the pylon assembly began to separate in the forward flange of the aft bulkhead.

3. The structural separation of the pylon was caused by a complete failure of the forward flange of the aft bulkhead after its residual strength had been critically reduced by the fracture and subsequent service life.

4. The overload fracture and fatigue cracking on the pylon aft bulkhead's upper flange were the only preexisting damage on the bulkhead. The length of the overload fracture and fatigue cracking was about 13 inches. The fracture was caused by an upward movement of the aft end of the pylon which brought the upper flange and its fasteners into contact with the wing clevis.

5. The pylon to wing attach hardware was properly installed at all attachment points.

6. All electrical power to the No. 1 a.c. generator bus and No. 1 d.c. bus was lost after the pylon separated. The captain's flight director instrument, the stall warning system, and the slat disagreement warning light systems were rendered inoperative. Power to these buses was never restored.

7. The No. 1 hydraulic system was lost when the pylon separated. Hydraulic systems No. 2 and No. 3 operated at their full capability throughout the flight. Except for spoiler panels No. 2 and No. 4 on each wing, all flight controls were operating.

8. The hydraulic lines and followup cables of the drive actuator for the left wing's outboard leading edge slat were severed by the separation of the pylon and the left wing's outboard slats retracted during climbout. The retraction of the slats caused an asymmetric stall and subsequent loss of control of the aircraft.

9. The flightcrew could not see the wings and engines from the cockpit. Because of the loss of the slat disagreement light and the stall warning system, the flightcrew would not have received an electronic warning of either the slat asymmetry or the stall. The loss of the warning systems created a situation which afforded the flightcrew an inadequate opportunity to recognize and prevent the ensuing stall of the aircraft.

10. The flightcrew flew the aircraft in accordance with the prescribed emergency procedure which called for the climbout to be flown at V_2 speed. V_2 speed was 6 KIAS below the stall speed for the left wing. The deceleration to V_2 speed caused the aircraft to stall. The start of the left roll was the only warning the pilot had of the onset of the stall.

11. The pylon was damaged during maintenance performed on the accident aircraft at American Airline's Maintenance Facility at Tulsa, Oklahoma, on March 29 and 30, 1979.

12. The design of the aft bulkhead made the flange vulnerable to damage when the pylon was being separated or attached.

13. American Airlines engineering personnel developed an ECO to remove and reinstall the pylon and engine as a single unit. The ECO directed that the combined engine and pylon assembly be supported, lowered, and raised by a forklift. American Airlines engineering personnel did not perform an adequate evaluation of either the capability of the forklift to provide the required precision for the task, or the degree of difficulty involved in placing the lift properly, or the consequences of placing the lift improperly. The ECO did not emphasize the precision required to place the forklift properly.

14. The FAA does not approve the carriers' maintenance procedures, and a carrier has the right to change its maintenance procedures without FAA approval.

15. American Airlines personnel removed the aft bulkhead's bolt and bushing before removing the forward bulkhead attach fittings. This permitted the forward bulkhead to act as a pivot. Any advertent or inadvertent loss of forklift support to the engine and pylon assembly would produce an upward movement at the aft bulkhead's upper flange and bring it into contact with the wing clevis.

16. American Airlines maintenance personnel did not report formally to their maintenance engineering staff either their deviation from the removal sequence contained in the ECO or the difficulties they had encountered in accomplishing the ECO's procedures.

17. American Airline's engineering personnel did not perform a thorough evaluation of all aspects of the maintenance procedures before they formulated the ECO. The engineering and supervisory personnel did not monitor the performance of the ECO to insure either that it was being accomplished properly or if their maintenance personnel were encountering unforeseen difficulties in performing the assigned tasks.

18. The nine situations in which damage was sustained and cracks were found on the upper flange were limited to those operations wherein the engine and pylon assembly was supported by a forklift.

19. On December 19, 1978, and February 22, 1979, Continental Airlines maintenance personnel damaged aft bulkhead upper flanges in a manner similar to the damage noted on the accident aircraft. The carrier classified the cause of the damage as maintenance error. Neither the air carrier nor the manufacturer interpreted the regulation to require that it further investigate or report the damages to the FAA.

20. The original certification's fatigue-damage assessment was in conformance with the existing requirements.

21. The design of the stall warning system lacked sufficient redundancy; there was only one stickshaker motor; and further, the design of the system did not provide for crossover information to the left and right stall warning computers from the applicable leading edge slat sensors on the opposite side of the aircraft.

22. The design of the leading edge slat system did not include positive mechanical locking devices to prevent movement of the slats by external loads following a failure of the primary controls. Certification was based upon acceptable flight characteristics with an asymmetrical leading edge slat condition.

23. At the time of DC-10 certification, the structural separation of an engine pylon was not considered. Thus, multiple failures of other systems resulting from this single event was not considered.

3.2 Probable Cause

The National Transportation Safety Board determines that the probable cause of this accident was the asymmetrical stall and the ensuing roll of the aircraft because of the uncommanded retraction of the left wing outboard leading edge slats and the loss of stall warning and slat disagreement indication systems resulting from maintenance-induced damage leading to the separation of the No. 1 engine and pylon assembly at a critical point during takeoff. The separation resulted from damage by improper maintenance procedures which led to failure of the pylon structure.

Contributing to the cause of the accident were the vulnerability of the design of the pylon attach points to maintenance damage; the vulnerability of the design of the leading edge slat system to the damage which produced asymmetry; deficiencies in Federal Aviation Administration surveillance and reporting systems which failed to detect and prevent the use of improper maintenance procedures; deficiencies in the practices and communications among the operators, the manufacturer, and the FAA which failed to determine and disseminate the particulars regarding previous maintenance damage incidents; and the intolerance of prescribed operational procedures to this unique emergency.

4. Safety Recommendations

As a result of this accident, the National Transportation Safety Board has recommended that the Federal Aviation Administration:

Issue immediately an emergency Airworthiness Directive to inspect all pylon attach points on all DC-10 aircraft by approved inspection methods. (Class I, Urgent Action) (A-79-41)

Issue a telegraphic Airworthiness Directive to require an immediate inspection of all DC-10 aircraft in which an engine pylon assembly has been removed and reinstalled for damage to the wing-mounted pylon aft bulkhead, including its forward flange and the attaching spar web and fasteners. Require removal of any sealant which may hide a crack in the flange area and employ eddy-current or other approved techniques to ensure detection of such damage. (Class I, Urgent Action) (A-79-45)

Issue a Maintenance Alert Bulletin directing FAA Maintenance inspectors to contact their assigned carriers and advise them to immediately discontinue the practice of lowering and raising the pylon with the engine still attached. Carriers should adhere to the procedure recommended by the Douglas Aircraft Company Service Bulletin which include removing the engine from the pylon before removing the pylon from the wing. (Class I, Urgent Action) (A-79-46)

Issue a Maintenance Alert Bulletin to U.S. certificated air carriers, and notify States that have regulatory responsibilities over foreign air carriers operating DC-10 aircraft, to require appropriate structural inspections of the engine pylons following engine failures involving significant imbalance conditions or severe side loads. (Class I, Urgent Action) (A-79-52)

Incorporate in type certification procedures consideration of:

(a) Factors which affect maintainability, such as accessibility for inspection, positive or redundant retention of connecting hardware and the clearances of interconnecting parts in the design of critical structural elements; and

(b) Possible failure combinations which can result from primary structural damage in areas through which essential systems are routed. (Class II--Priority Action) (A-79-98)

Insure that the design of transport category aircraft provides positive protection against asymmetry of lift devices during critical phases of flight; or, if certification is based upon

demonstrated controllability of the aircraft under condition of asymmetry, insure that asymmetric warning systems, stall warning systems, or other critical systems needed to provide the pilot with information essential to safe flight are completely redundant. (Class II--Priority Action) (A-79-99)

Initiate and continue strict and comprehensive surveillance efforts in the following areas:

(a) Manufacturer's quality control programs to assure full compliance with approved manufacturing and process specifications; and

(b) Manufacturer's service difficulty and service information collection and dissemination systems to assure that all reported service problems are properly analyzed and disseminated to users of the equipment, and that approriate and timely corrective actions are effected. This program should include full review and specific FAA approval of service bulletins which may affect safety of flight. (Class II--Priority Action) (A-79-100)

Assure that the Maintenance Review Board fully considers the following elements when it approves an Airline/Manufacturer Maintenance Program:

(a) Hazard analysis of maintenance procedures which involve removal, installation, or work in the vicinity of structurally significant 1/ components in order to identify and eliminate the risk of damage to those components;

(b) Special inspections of structurally significant components following maintenance affecting these components; and

(c) The appropriateness of permitting "On Condition" maintenance and, in particular, the validity of sampling inspection as it relates to the detection of damage which could result from undetected flaws or damage to structurally significant elements during manufacture or maintenance. (Class II--Priority Action) (A-79-101)

Require that air carrier maintenance facilities and other designated repair stations:

1/ Structurally significant items as defined in Appendix 1 of Advisory Circular 120-17A-"Maintenance Control By Reliability Methods."

(a) Make a hazard analysis evaluation of proposed maintenance procedures which deviate from those in the manufacturer's manual and which involve removal, installation, or work in the vicinity of structurally significant components; and

(b) Submit proposed procedures and analysis to the appropriate representative of the Administrator, FAA, for approval. (Class II--Priority Action) (A-79-102)

Revise 14 CFR 121.707 to more clearly define "major" and "minor" repair categories to insure that the reporting requirement will include any repair of damage to a component identified as "structurally significant." (Class II--Priority Action) (A-79-103)

Expand the scope of surveillance of air carrier maintenance by:

(a) Revising 14 CFR 121 to require that operators investigate and report to a representative of the Administrator the circumstances of any incident wherein damage is inflicted upon a component identified as "structurally significant" regardless of the phase of flight, ground operation, or maintenance in which the incident occurred; and

(b) Requiring that damage reports be evaluated by appropriate FAA personnel to determine whether the damage cause is indicative of an unsafe practice and assuring that proper actions are taken to disseminate relevant safety information to other operators and maintenance facilities. (Class II--Priority Action) (A-79-104)

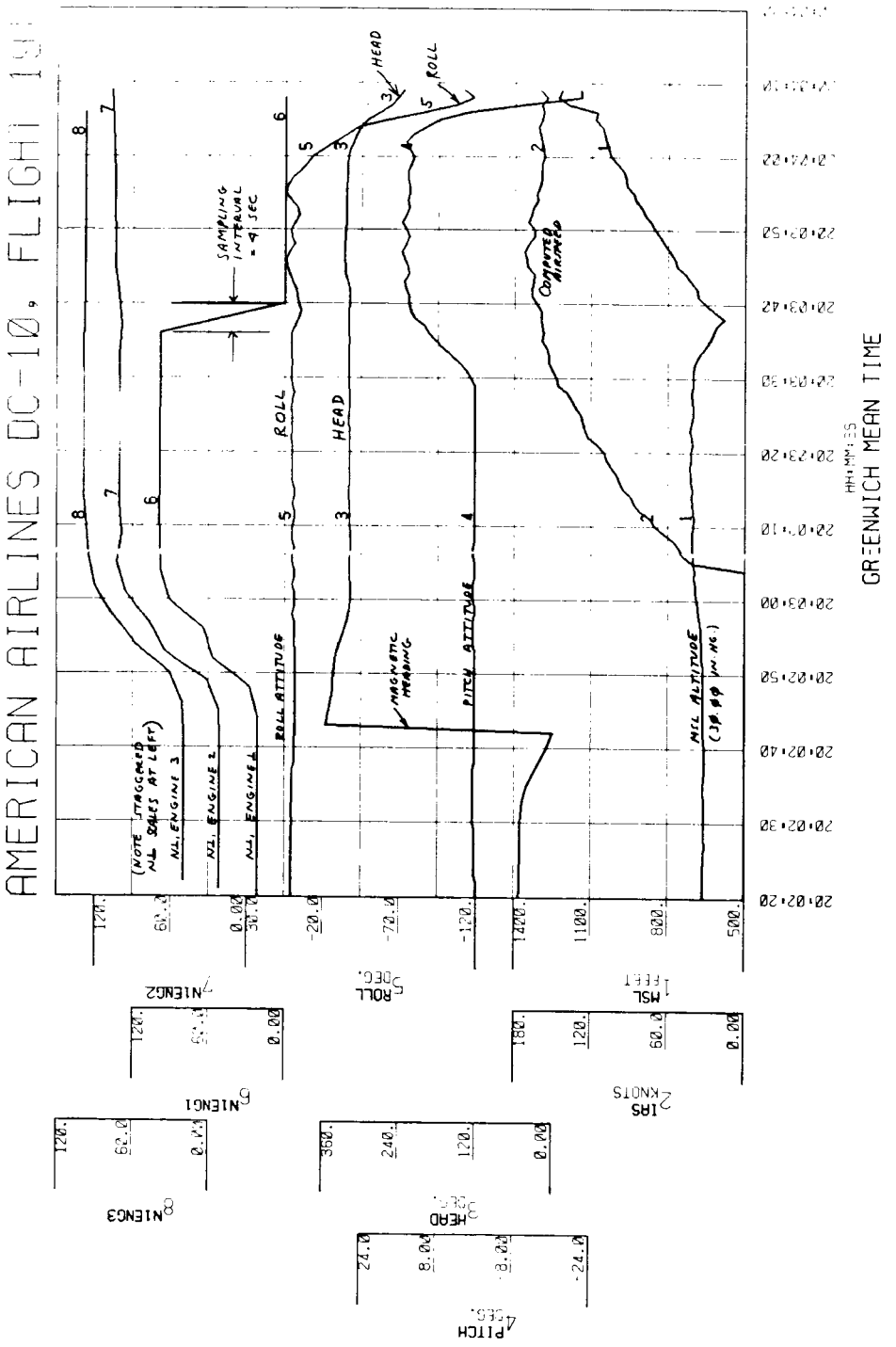
Revise operational procedures and instrumentation to increase stall margin during secondary emergencies by:

(a) Evaluating the takeoff-climb airspeed schedules prescribed for an engine failure to determine whether a continued climb at speeds attained in excess of V_2, up to V_2 + 10 knots, is an acceptable means of increasing stall margin without significantly degrading obstacle clearance.

(b) Amending applicable regulations and approved flight manuals to prescribe optimum takeoff-climb airspeed schedules; and

(c) Evaluating and modifying as necessary the logic of flight director systems to insure that pitch commands in the takeoff and go-around modes correspond to optimum airspeed schedules as determined by (a) and (b) above. (Class II-Priority Action) (A-79-105)

APPENDIX H


AMERICAN AIRLINES DC-10, FLIGHT 19
(NOTE STAGGERED N1 SCALES AT LEFT)
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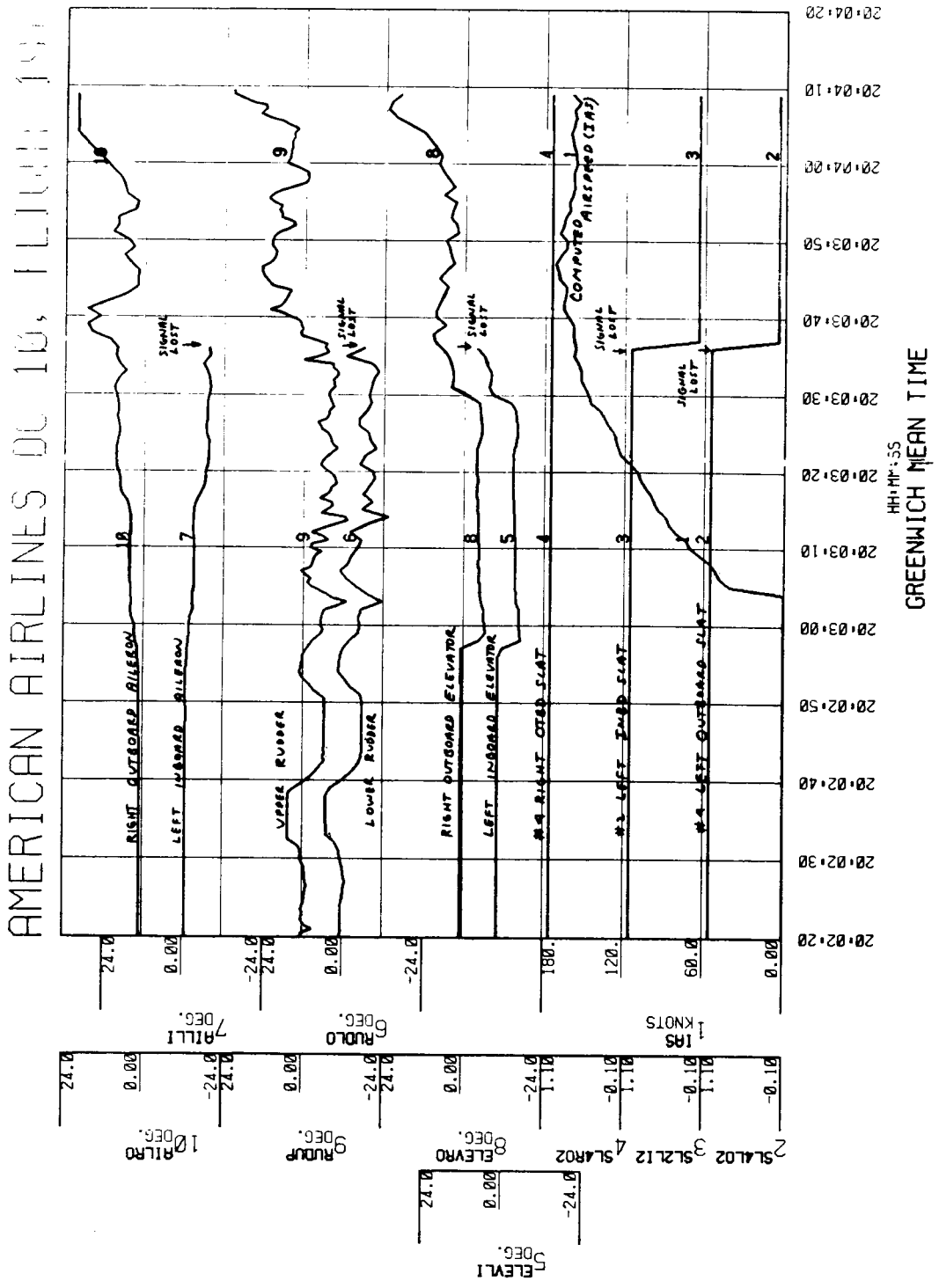
N1, ENGINE 2
N1, ENGINE 1
ROLL ATTITUDE
HEAD
ROLL
SAMPLING INTERVAL = 4 SEC
MAGNETIC HEADING
PITCH ATTITUDE
N1 ALTITUDE (28.89 IN HG)
HEAD
ROLL
GREENWICH MEAN TIME
HH:MM:SS
20:02:20
20:02:30
20:02:40
20:02:50
20:03:00
20:03:10
20:23:20
20:23:30
20:23:40
20:23:50
20:24:00
20:24:10
1 MSL FEET
2 IAS KNOTS
3 ROLL DEG.
4 PITCH DEG.
5 HEAD DEG.
6 N1 ENG3
7 N1 ENG2
8 N1 ENG1


AMERICAN AIRLINES DC 10, FLIGHT 191
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LEFT INBOARD AILERON
UPPER RUDDER
LOWER RUDDER
RIGHT OUTBOARD ELEVATOR
LEFT INBOARD ELEVATOR
#4 RIGHT OTBD FLAT
#3 LEFT INBD FLAT
#4 LEFT OUTBOARD FLAT
COMPUTED AIRSPEED (IAS)
SIGNAL LOST
GREENWICH MEAN TIME
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20:03:20
20:03:30
20:03:40
20:03:50
20:04:00
20:04:10
20:04:20
IAS KNOTS
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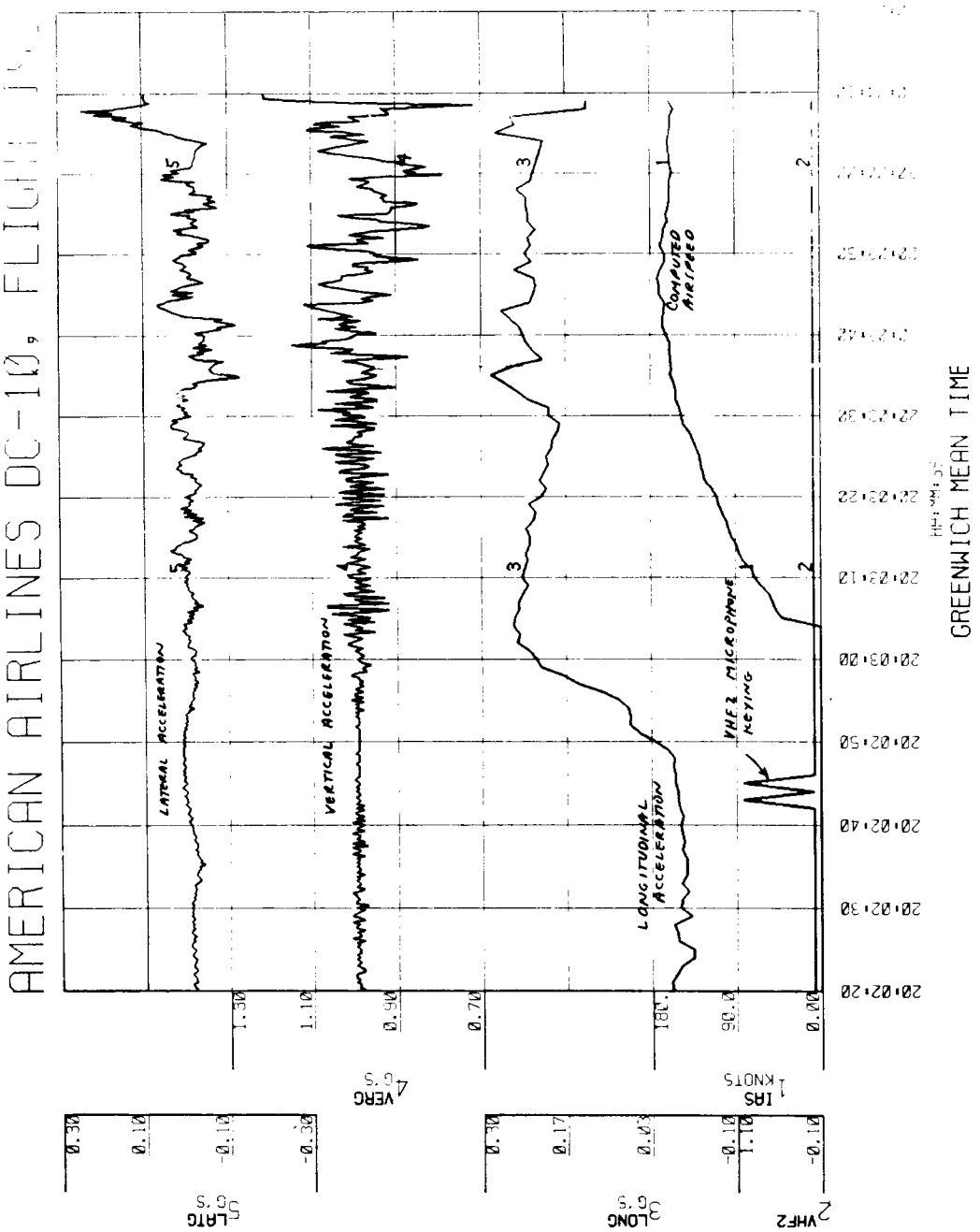
AILLI DEG.
ELEVLI DEG.
ELEVRO DEG.
RUDUP DEG.
AILRO DEG.
SLAT02
SLAT12
SLAT02



ICAO Note: Appendices A - G inclusive to this Report were not reproduced.

ICAO Ref: 131/79

ICAO TECHNICAL PUBLICATIONS

The following summary gives the status, and also describes in general terms the contents of the various series of technical publications issued by the International Civil Aviation Organization. It does not include specialized publications that do not fall specifically within one of the series, such as the Aeronautical Chart Catalogue *or the* Meteorological Tables for International Air Navigation.

International Standards and Recommended Practices are adopted by the Council in accordance with Articles 54, 37 and 90 of the Convention on International Civil Aviation and are designated, for convenience, as Annexes to the Convention. The uniform application by Contracting States of the specifications contained in the International Standards is recognized as necessary for the safety or regularity of international air navigation while the uniform application of the specifications in the Recommended Practices is regarded as desirable in the interest of safety, regularity or efficiency of international air navigation. Knowledge of any differences between the national regulations or practices of a State and those established by an International Standard is essential to the safety or regularity of international air navigation. In the event of non-compliance with an International Standard, a State has, in fact, an obligation, under Article 38 of the Convention, to notify the Council of any differences. Knowledge of differences from Recommended Practices may also be important for the safety of air navigation and, although the Convention does not impose any obligation with regard thereto, the Council has invited Contracting States to notify such differences in addition to those relating to International Standards.

Procedures for Air Navigation Services (PANS) are approved by the Council for world-wide application. They contain, for the most part, operating procedures regarded as not yet having attained a sufficient degree of maturity for adoption as International Standards and Recommended Practices, as well as material of a more permanent character which is considered too detailed for incorporation in an Annex, or is susceptible to frequent amendment, for which the processes of the Convention would be too cumbersome.

Regional Supplementary Procedures (SUPPS) have a status similar to that of PANS in that they are approved by the Council, but only for application in the respective regions. They are prepared in consolidated form, since certain of the procedures apply to overlapping regions or are common to two or more regions.

The following publications are prepared by authority of the Secretary General in accordance with the principles and policies approved by the Council.

Technical Manuals provide guidance and information in amplification of the International Standards, Recommended Practices and PANS, the implementation of which they are designed to facilitate.

Air Navigation Plans detail requirements for facilities and services for international air navigation in the respective ICAO Air Navigation Regions. They are prepared on the authority of the Secretary General on the basis of recommendations of regional air navigation meetings and of the Council action thereon. The plans are amended periodically to reflect changes in requirements and in the status of implementation of the recommended facilities and services.

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